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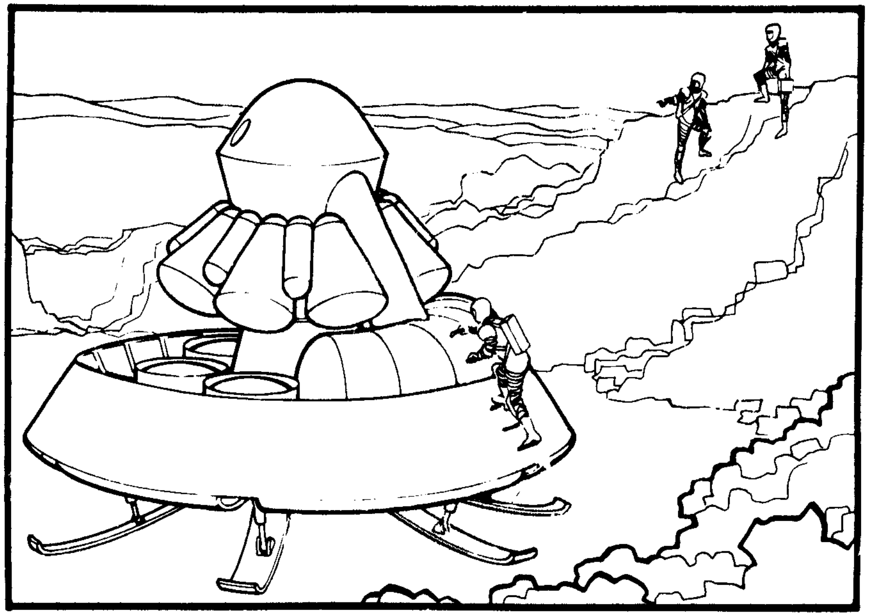

DEFINITION OF EXPERIMENTAL TESTS FOR A MANNED MARS EXCURSION MODULE

FINAL REPORT
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SPACE DIVISION
NORTH AMERICAN ROCKWELL CORPORATION

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DEFINITION OF EXPERIMENTAL TESTS FOR A MANNED MARS EXCURSION MODULE FINAL BRIEFING

NAS9-6464

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SPACE DIVISION
NORTH AMERICAN ROCKWELL CORPORATION

FOREWORD

This briefing brochure presents a synopsis of the study and is submitted in accordance with Article VIII, Paragraph B of Contract NAS9-6464, "Definition of Experimental Tests for a Manned Mars Excursion Module," issued by the Manned Spacecraft Center, National Aeronautics and Space Administration located in Houston, Texas. The data were generated between October 1966 and August 1967 and are presented in four volumes:

Volume I	-	Summary	(SD 67-755-1)
Volume II	-	Design	(SD 67-755-2)
Volume III	-	Test Program	(SD 67-755-3)
Volume IV	-	Briefing Brochure	(SD 67-755-4)

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INTRODUCTION

Consideration is being given to the design and planning of Mars exploration missions in the 1980's. An important element in the accomplishment of these missions is the Mars Excursion Module (MEM) which will provide the means for a team of astronauts to land and take off from the planet. A study conducted in 1963 indicated the potential complexity and extended duration requirements of such a vehicle. Since completion of these preliminary investigations, data derived from the Mariner IV flyby and further astronomical observations indicate that the Martian atmosphere is more tenuous than was initially assumed (e.g., nominal surface pressure estimates of 15 to 130 mb in 1963 have been reduced to 5 to 10 mb). Consequently, the applicability of the earlier designs, particularly the retardation and landing subsystem concepts, required review and updating. Furthermore, to provide data for the formulation of a total system development program for the mission, it appeared both appropriate and timely to consider the definition of a MEM development and test program in order to generate preliminary estimates of the schedule and funding level requirements for a manned landing on Mars in the 1980's.

This briefing, given in three parts (i.e., Summary, Design, and Test Program), presents the results of a one-year study to develope a preliminary conceptual design and determine the requirements and formulate a test program for the MEM. The study, begun in October 1966, was conducted by the Space Division of North American Rockwell for the NASA Manned Spacecraft Center under Contract NAS9-6464.

STUDY OBJECTIVES

The objectives of the study were to (1) perform a conceptual design study of a MEM; (2) define a test and qualification program which ultimately would demonstrate system and mission feasibility, including the requirements for vehicles to perform such tests; (3) consider the requirements for and feasibility of designing an unmanned Mars test vehicle which later could be converted into a manned MEM with minimum modifications; (4) determine the resource requirements, including costs and schedules, to qualify a MEM; and (5) identify the technological requirements to accomplish the test program and Mars landing.

Approximately 90 percent of the activity was apportioned equally between the first two objectives, and the remaining 10 percent was allocated to the schedule and resource requirements evaluation. Those areas wherein additional or future research and development may be required were noted and evaluated throughout the study.

STUDY OBJECTIVES

PERFORM CONCEPTUAL DESIGN

DETERMINE TEST REQUIREMENTS & FORMULATE A MEM TEST PROGRAM

EVALUATE FEASIBILITY & DESIRABILITY OF UNMANNED MARS FLIGHTS

DETERMINE SCHEDULE & RESOURCE REQUIREMENTS

IDENTIFY TECHNOLOGY IMPLICATIONS

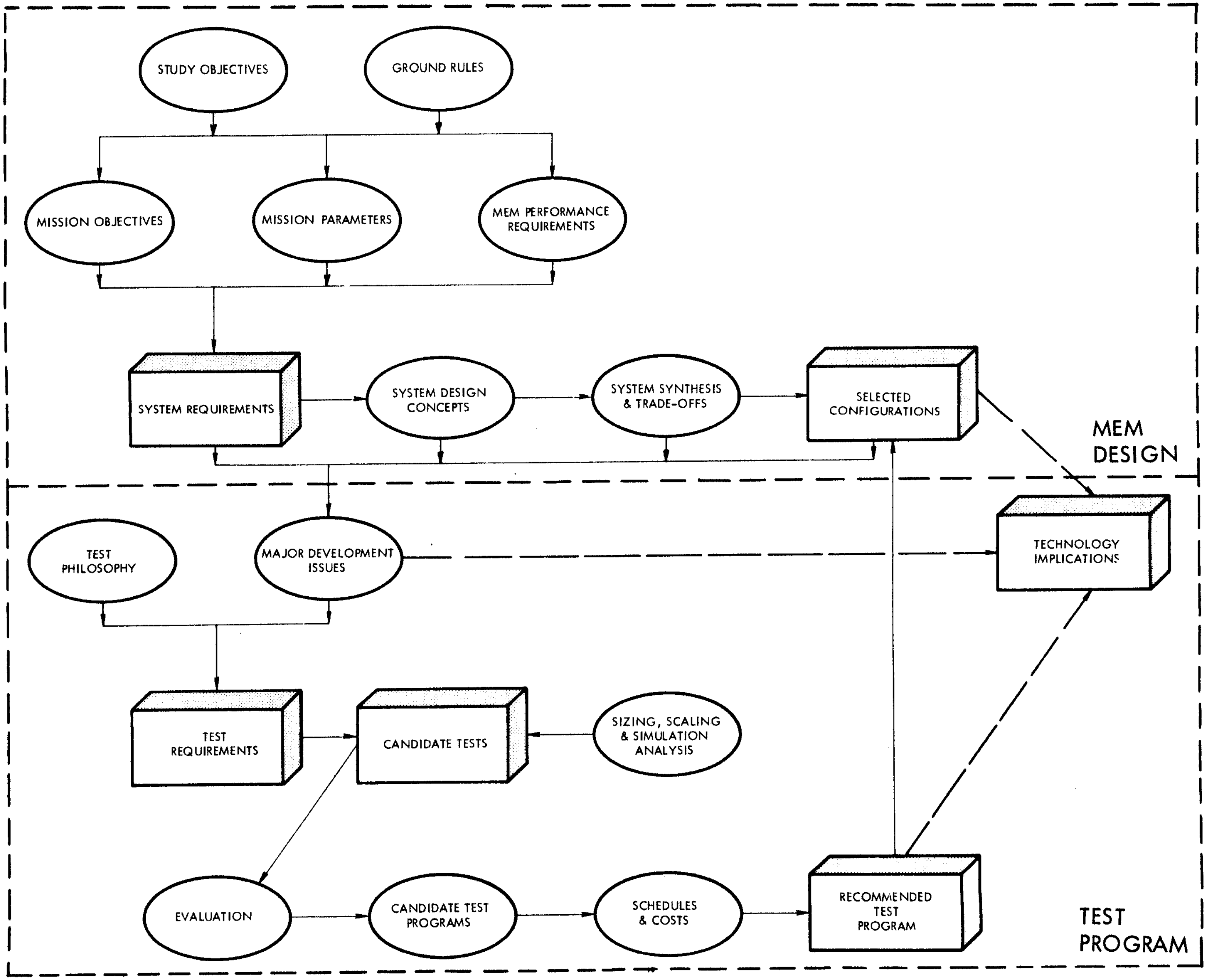
STUDY LOGIC DIAGRAM

The first phase of the study was concerned with the MEM design. The system requirements were evaluated and the MEM design criteria established. Consideration was given to the scientific mission objectives (both surface and orbital) and their anticipated effect on the selection of a landing site; compatibility of the desirable landing site locations with mission-imposed constraints (e.g., energy requirements, approach and departure velocity requirements, abort, orbital parameters, etc.); performance requirements for deorbit, entry, retardation and landing, ascent and rendezvous, including surface abort penalties; and the development of a nominal mission profile.

MEM design concepts were generated for both low-lift (Apollo shape - $L/D \approx 0.5$) and moderate lift ($L/D \approx 1.0$) configurations. The major subsystems (i.e., propulsion, thermostructure, and retardation and landing) and other subsystems (i.e., environmental control and life support, guidance and control, communications and power) were evaluated discretely and obvious selections made where-ever possible based on requirements and tradeoffs. Minimum volume and weight ascent capsule and laboratory compartments were first configured followed by conceptual designs of the ascent propulsion system, retardation and landing systems, entry heat shield and structure, and deorbit propulsion. Low and moderate L/D configurations were compared and the effects of crew size, stay time, retardation and landing mode (i.e., ballutes and retro-propulsion versus retro-propulsion only, hover time, propellant performance, etc.) were analyzed. The more attractive designs were segregated on the basis of weight, system compatibility, and growth potential. The selected concepts then were introduced as candidates in the second phase of the study so that the cost and schedule risk in accomplishing the requisite development and qualification test program could be evaluated.

Given the system requirements as well as the candidate configurations, test requirements were established in the second phase of the study based upon the major system and subsystem development issues and a test philosophy predicated upon the experience obtained on other programs. A list of candidate Earth tests, unmanned flight tests, and manned flights then was prepared considering applicable sizing, scaling, and similitude relationships. The candidate tests were evaluated and several potential test programs were evolved. These test programs were then examined in light of schedule, cost, and technology implications and weighed against the anticipated yield of data and increased confidence. Final review of data resulted in the selection of a recommended program and configuration.

STUDY LOGIC DIAGRAM



SUMMARY BRIEFING OUTLINE

The summary briefing has been organized to present the highlights of the study results and, consequently, each of the major areas of investigation will be discussed briefly. The reader is referred to the body of the technical briefing and the several volumes of the final reports for details.

Included in the summary briefing are:

a. The recommended design

b. The recommended test and development program

c. Schedule and cost implications

d. Technological implications

SUMMARY BRIEFING OUTLINE

GROUND RULES

MISSION PROFILE

RECOMMENDED DESIGN

TEST PROGRAM

SCHEDULE AND COSTS

CONCLUSIONS

STUDY GROUND RULES

The following study ground rules were suggested by the NASA COR:

1. For purposes of this investigation, the first opportunity considered for a manned landing on Mars was the 1982 opportunity.

2. The test program to demonstrate and qualify the MEM was to be accomplished in the 1971-1978 time period. It was preferred that major flight tests be made with single Saturn V launches. The use of uprated Saturn V's for multiple launches could be considered. The requirements for the uprated booster, if necessary, were to be specified.

3. The atmospheric models to be used were the JPL VM-7 and VM-8 models and the NASA upper and mean density models. The former two have surface pressures of 5 mb; the latter pressures are 10 and 8 mb, respectively. The models also differ in composition, temperature, and scale height. The MEM was to be designed for the overall "worst case" atmosphere, as identified from the four models given. The sensitivity to the other models was to be examined.

4. Configurations to be studied included both low L/D Apollo-types and moderate L/D lifting body configurations. For purposes of identifying the performance requirements, these configurations were assumed to have L/D values of 0.5 and 1.0.

5. Crews of two and four men and Mars surface stay times of 4 and 30 days were considered. The two combinations of particular interest were a minimum mission consisting of two men for a four-day stay time and a four-man 30-day mission.

6. The scientific payloads were identified by NASA as a 4500-pound (2040 kg) payload to be taken down to and left on the surface and a 3000-pound (136 kg) payload to be returned to the spacecraft in orbit.

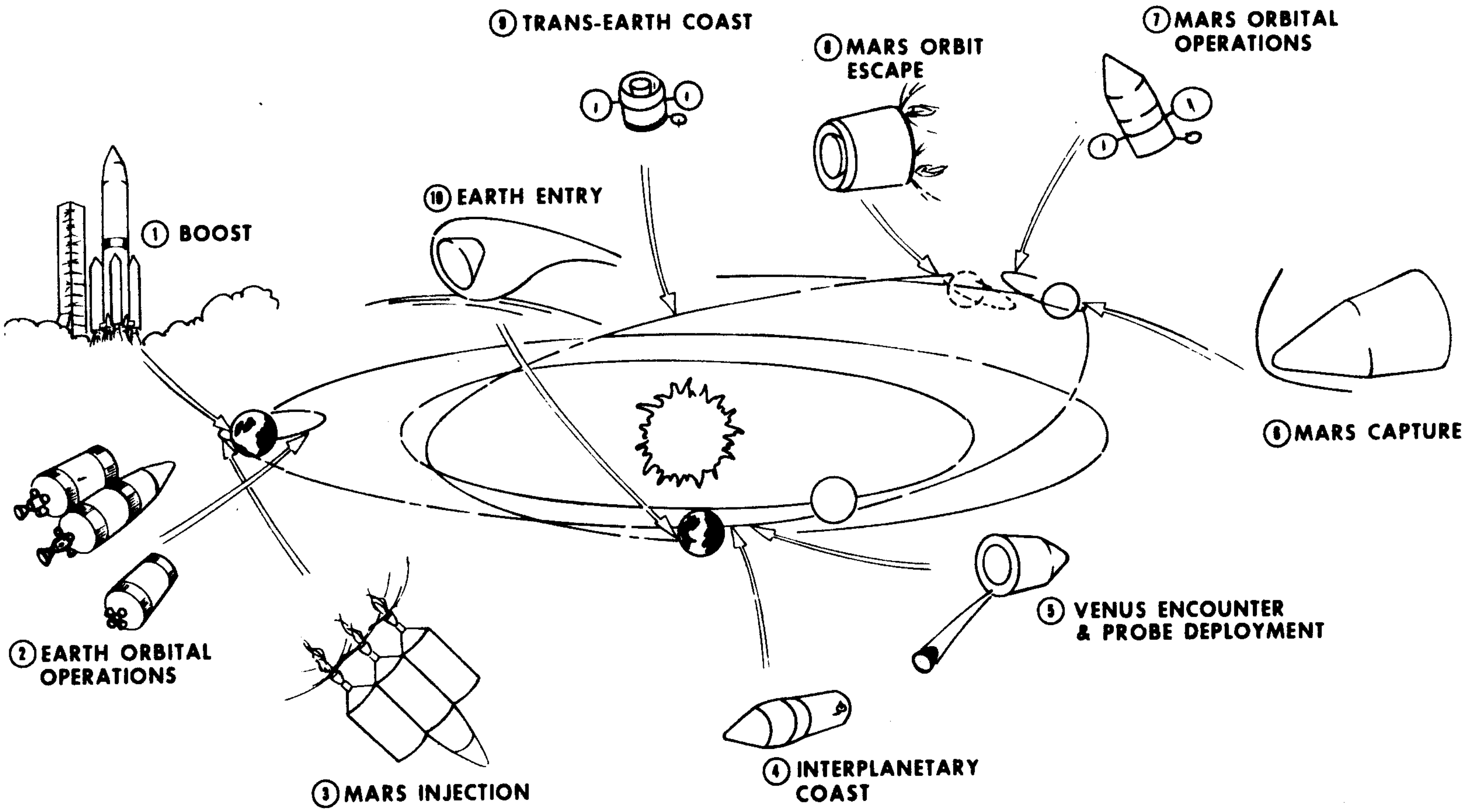
STUDY GROUND RULES

FIRST MARS LANDING	1982 EARLIEST OPPORTUNITY CONSIDERED
TEST & QUALIFICATION PROGRAM	1971 TO 1978 SINGLE SATURN V LAUNCH PREFERRED
ATMOSPHERE MODELS	JPL VM-7 & VM-8 , NASA UPPER & MEAN DENSITY MODELS
CONFIGURATIONS	LOW (0.5) & MODERATE (1.0) L/D COMPATIBLE WITH 33 FEET (10 M) LAUNCH VEHICLE
CREW SIZE/STAY TIME	4-MAN/30-DAY 2-MAN/4-DAY
SCIENTIFIC PAYLOAD	4500 LB (2020 KG) TO SURFACE 300 LB (136 KG) TO ORBIT

MARS AEROBRAKER MISSION PROFILE

The MEM would be transported to Mars by an orbiter spacecraft. This spacecraft, which might be an aerobraker or retrobraker, would be launched into Earth orbit by multiple launches and assembled and checked out over a period of several weeks. The MEM could be launched (unmanned) with the spacecraft and not be manned (except for checkout operations) until it is in Mars orbit. An aerobraker spacecraft is illustrated in a Venus swingby mode. After Mars orbital capture, the MEM would separate, land, and later return to the orbiting spacecraft. After rendezvous, docking, and crew transfer, the MEM would be abandoned in Mars orbit and the spacecraft injected back to Earth. An Earth entry module would be employed for reentry at hyperbolic speeds. The mission duration would be 400 to 700 days, depending on the mode selected.

MARS AEROBRAKER MISSION PROFILE
VENUS SWINGBY



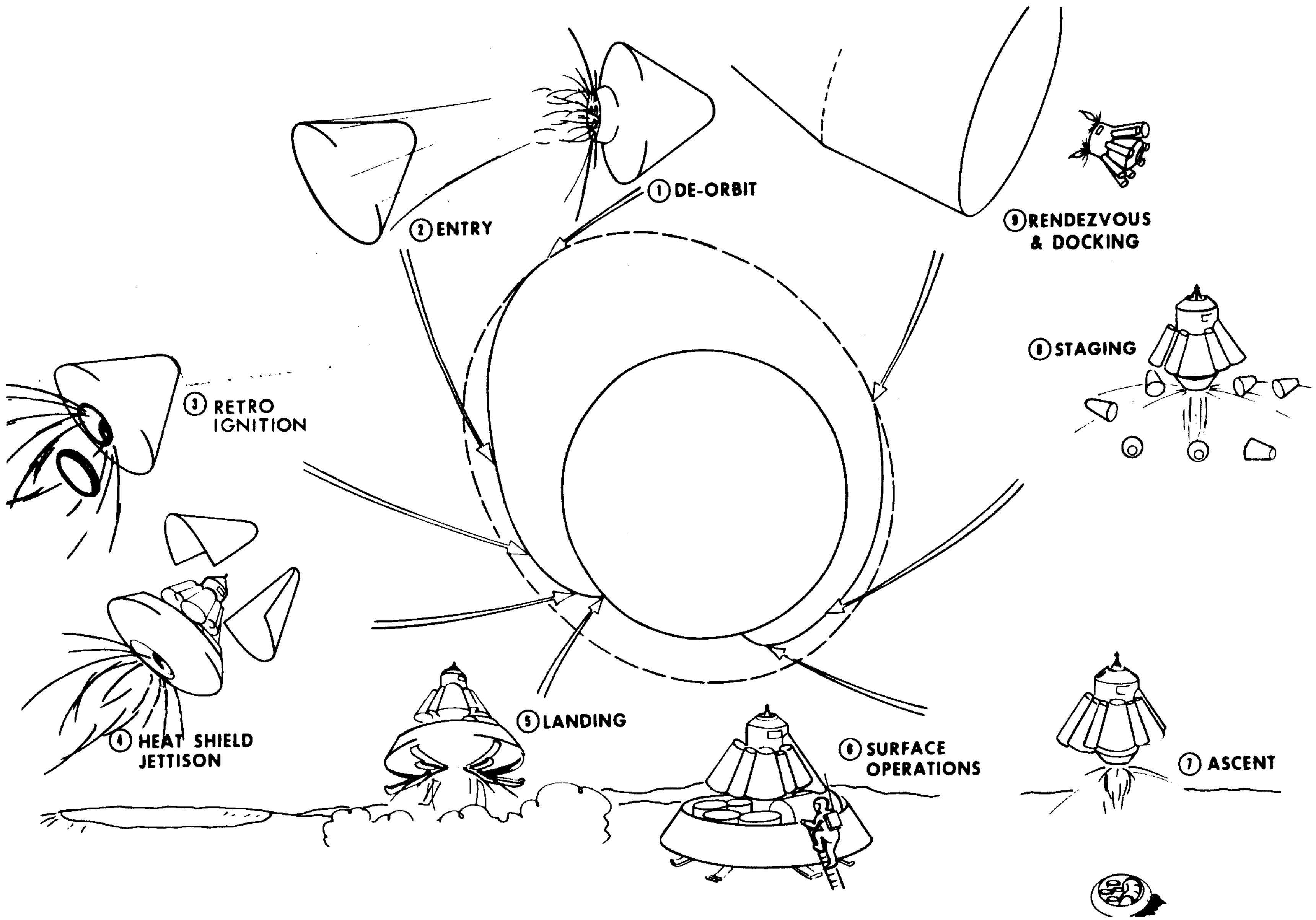
MEM MISSION PROFILE

The MEM would be passive and unmanned during the Earth orbital operations and interplanetary transit phases except for scheduled checkout and maintenance operations. After the Mars capture orbit has been achieved, the subsystems would be checked out and activated, and the MEM manned and separated from the spacecraft. Low thrust de-orbit motors would be fired at a predetermined time and position to effect entry and landing in a preselected landing area. Entry generally would occur with the lift vector up; roll control would be employed for minor navigational adjustments. To decelerate the MEM, retropropulsive thrust would be initiated at the equilibrium velocity of about 3000 fps (0.9 km/sec) and applied so that the vertical component balances the difference between weight and lift. Portions of the heat shield would be jettisoned to reduce weight. Touchdown would occur after a short hover period over the final landing site. Until touchdown, the crew would occupy the control cabin atop the vehicle.

A laboratory and living quarters, connected to the control cabin by a tunnel and airlock, would be provided for surface operations. At lift-off, much of the equipment and structure would be left on the surface; propulsion tankage would be staged during ascent. Normally, the MEM would ascend to an intermediate phasing orbit and, after appropriate phasing with the spacecraft orbit, effect rendezvous and docking. After the crew and scientific payload are transferred to the spacecraft, the MEM would be abandoned in Mars orbit.

Abort capability exists before entry, before landing, and on the Mars surface; there is no abort capability during entry. The most critical abort requirement is imposed just before touchdown when the ascent stage must be separated, the ascent engine ignited, and a turn-around maneuver performed to correctly orient the thrust vector.

MEM MISSION PROFILE



DESIGN REQUIREMENTS

The design missions selected for the MEM included a nominal 4-man/30-day mission and a 2-man/4-day mission. Spacecraft parking orbits considered varied from a 270 nm (500 km) circular orbit, which is the most simple operationally and results in the lowest entry velocity and ascent ΔV requirements, to a 162 by 36,100 nm (300 by 66,900 km) elliptical orbit ($e = 0.9$) which results in minimum combined system (i.e., spacecraft/MEM) weight and maximum entry velocities and ascent ΔV requirements. The orbital inclination and the landing site location have only minor effects on the MEM design requirements. The VM-7 atmosphere was selected for the design because it imposes the most severe system requirements; sensitivities to the other atmospheres were considered.

Nominal lift-to-drag ratios of 0.5 and 1.0 were considered. To assure compatibility with the 33-foot (10 m) diameter Saturn V and a concurrent aerobraker study, the MEM diameter was constrained to 31.5 feet (9.6 m).

If aerobraking is employed to accomplish Mars orbital capture, the loads may approach 10 g (Earth); all other loads were limited to less than 5 g. Entry velocities will vary from 11,350 fps (3.46 km/sec) for the low circular orbit mission to 16,000 fps (4.88 km/sec) for the elliptical orbit ($e = 0.9$) mission.

The deorbit and retardation propulsion requirements are essentially the same over the range of missions selected. A ΔV of approximately 660 fps (0.2 km/sec) is required to deorbit from the low circular orbit. Although considerably lower ΔV's are required for deorbit at the apoapsis of the elliptic orbit, this excess capability will provide operational flexibility for deorbit at other true anomalies. The retardation ΔV varies as a function of $m/C_L A$ and includes an allowance for a two-minute hover period. Thrust is initiated at the higher thrust level and throttled back at touchdown. The ascent ΔV is a function of the parking orbit and varies from 16,000 fps (4.88 km/sec) for the low circular and to 20,350 fps (6.20 km/sec) for highly elliptical orbits. The rendezvous propellant allocation was equivalent to 300 fps (0.1 km/sec).

DESIGN REQUIREMENTS

MISSION

4 MEN/30 DAYS AND 2 MEN/4 DAYS STAY TIME
270 NM CIRCULAR TO 162 X 36,100 NM ORBIT
VM-7 ATMOSPHERE

CONFIGURATION

L/D = 0.5 & 1.0 : MAXIMUM DIAMETER $\leq$ 31.5 FT
LAUNCH, ENTRY, & IMPACT LOADS ~5g
(AEROBRAKING MANEUVER $\leq$ 10g)

ENTRY VELOCITY

FROM LOW CIRCULAR ORBIT ~ 11,000 FT/SEC
FROM MAXIMUM ELLIPTIC ORBIT $\gtrsim$ 16,000 FT/SEC

PROPULSION

PHASE	ΔV (FT/SEC)	$T/W_{\oplus}$
ORBIT	660	0.4
RETARDATION	3500 - 4750	1.5 - 0.15
ASCENT	16,000 - 20,350	1.0
RENDEZVOUS	300	(RCS)

RECOMMENDED DESIGN

A low L/D 30-foot (9.1 m) diameter Apollo-shape configuration was selected as the recommended design. This MEM would be capable of performing the maximum mission, i.e., landing 4 men for 30 days and returning them to a 162 by 36,100 nm (300 by 66,900 km) orbit (e = 0.9).

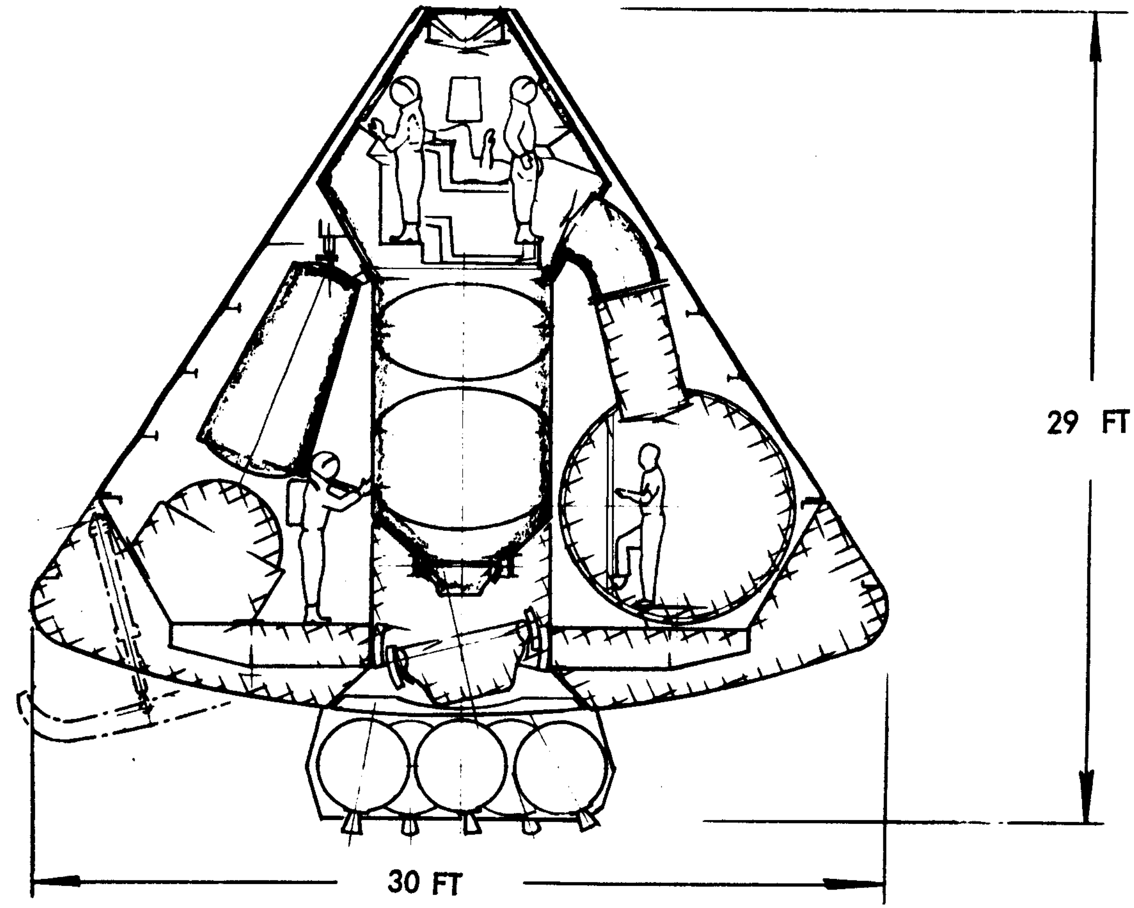
The deorbit motors are jettisoned after firing and the vehicle enters the atmosphere at an angle of attack of 147 degrees. The conical section of the heat shield and the plug over the retro-engine are jettisoned prior to igniting the retropropulsion engine. Although the crew compartment contains four couches, two of the crew will assume standing positions to land the vehicle. A six-legged landing gear, which is an integral part of the aft heat shield, attenuates the landing loads. (The landed stage of the vehicle is shown by the cross-hatching and the ascent stage by the shading.)

The crew quarters and laboratory are in the form of a toroidal section and are connected to the crew capsule by an airlock. Another airlock provides access to the Martian surface and interior of the vehicle for inspection and maintenance.

A 140,000-pound-(64,000 kg) thrust plug nozzle descent engine and a 35,000-pound (16,000 kg) ascent engine are arranged symmetrically. Both engines are gimballed; the descent engine is canted approximately 13 degrees to thrust through the off-set c.g. The c.g. off-set is achieved by arranging the eight spherical descent propellant tanks and eight first-stage ascent tanks asymetrically around the central structure. The two second-stage ascent tanks are mounted on the central thrust structure between the crew cabin and the ascent engine.

The vehicle weight would be 109,000 pounds (49,400 kg) at separation; the m/C_LA at entry of 7.4 slugs/ft^2 (1160 kg/m^2) is similar to that of the Apollo command module. To allow for possible growth in the vehicle or payload weight, a larger vehicle may be desirable. For example, given the same general arrangment and employing a 31.5 foot (9.6 m) base diameter, the gross weights can increase by approximately 50 percent and the highly elliptical orbit missions can still be accomplished.

RECOMMENDED DESIGN



CHARACTERISTICS

4 MAN/30 DAY

W_{GROSS} = 109,000 LBS

M/C_LA = 7.4 SLUGS/FT^2

ASCENT ΔV ($e = 0.9$) = 20,350 FPS

WEIGHT SUMMARY

The weight at separation of the recommended configuration (i. e., 30-ft diameter low L/D vehicle, 4-man 30-day elliptical orbit mission) will be approximately 109,000 lb (49,500 kg); the weight at entry will be 101,600 lb (46,100 kg).

The spacecraft weight is apportioned on a 64:37 ratio between the descent and ascent stage. Propellants comprise nearly half of the descent stage and more than half of the ascent stage weights. Inasmuch as the MEM weight ultimately is a function of the ascent capsule burnout weight, all items not essential to accomplishing the final rendezvous are left behind (e.g., structure, fuel cells, laboratory, etc.). Descent propellant weights include a two-minute hover capability.

WEIGHT SUMMARY

ITEM	LBS
ASCENT CAPSULE	(5260)
STRUCTURE	980
POWER	500
COMMUNICATION	210
GUIDANCE & CONTROL	225
ECLSS	1340
RCS	530
RETURN PAYLOAD	300
CREW	700
CONTINGENCY	475
STAGE II	(9430)
TANKS & SYSTEM	690
ENGINE	490
PROPELLANT	8250
STAGE I	(22,510)
TANKS & SYSTEM	1610
PROPELLANT	20,900
TOTAL ASCENT STAGE	37,200

ITEM	LBS
DESCENT STAGE	
JETTISONED STRUCTURE	4650
RETAINED STRUCTURE	6350
LAB STRUCTURE	1350
EPS	2240
COMMUNICATION	370
GUIDANCE & CONTROL	10
ECLSS	1630
RCS	2630
LANDING GEAR	2770
LANDING PAYLOAD	4200
CONTINGENCY	3070
TANKS & SYSTEM	2600
ENGINE	2020
PROPELLANT	30,500
TOTAL DESCENT STAGE	64,390
ENTRY WEIGHT	101,590
DEORBIT MOTORS	7400
GROSS MEM WEIGHT	108,990

HEAT SHIELD

Relevant parameters for the thermostructural subsystem are presented for:

1. Mars entry from a low circular orbit at 11,000 fps (3.4 km/sec).
2. Mars entry from a highly elliptical orbit at 16,000 fps (4.9 km/sec).
3. Entry at Earth orbital velocities of 25,040 fps (7.6 km/sec) for Earth orbital testing.

The typical weights shown include the ablator, insulation, and primary structure, but exclude such structural elements as frames and stringers. For entry from low circular orbits, only a small area around the stagnation point needs to be ablatively cooled and the heat shield weight is only 5500 pounds (2500 kg). For entry from highly elliptical orbits, the ablative coverage must be increased and resulting weight is 7300 pounds (3300 kg). For entry from Earth orbit, nearly all the vehicle must be covered with ablator and the weight is 8600 pounds (3900 kg).

A heat shield designed for entry from Earth orbit, although 1300 pounds heavier than one designed for entry from an elliptical Mars orbit, is recommended. The weight penalty is offset by the fact that such a heat shield is desensitized to the anticipated range of Mars entry conditions (e.g., velocities, uncertainties in atmospheric composition and density profiles); furthermore, the scope and cost of the MEM test program can be reduced significantly by eliminating the need for special suborbital heat shield qualification tests. (Peak heating rates in VM-8 are approximately 80 percent higher and the total heat loads are 30 percent lower than in VM-7. Intermediate heating rates and loads are encountered in the NASA upper and mean density models.)

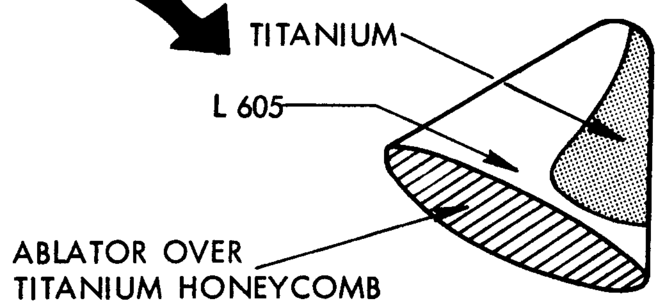
The recommended heat shield utilizes AVCOAT 5026-39 ablator over a titanium honeycomb substructure over the regions where the temperature during entry exceeds 1800 F (1250 K), L 605 cobalt-based superalloy where the temperatures are between 1000 and 1800 F (810 and 1250 K), and 6Al-4Va titanium where the temperatures are below 1000 F (810 K). The radiative portion of the structure consists of minimum gauge shingles supported by corrugated stiffeners.

HEAT SHIELD

PARAMETER	MARS ENTRY (VM-7)		EARTH ENTRY
ENTRY VELOCITY (FT/SEC)	~11,000	16,000	25,040
MAX HEATING RATE (BTU/FT^2-SEC)	19.4	55	102
TOTAL HEAT INPUT (BTU/FT^2)	4300	6800	19,600
HEAT SHIELD WEIGHT (LB)	5500	7300	8600

DESIGN HEATSHIELD FOR ENTRY FROM EARTH ORBIT

WEIGHT PENALTY ~1300 LB
DESENSITIZES MEM TO ENTRY CONDITIONS
REDUCED TEST PROGRAM



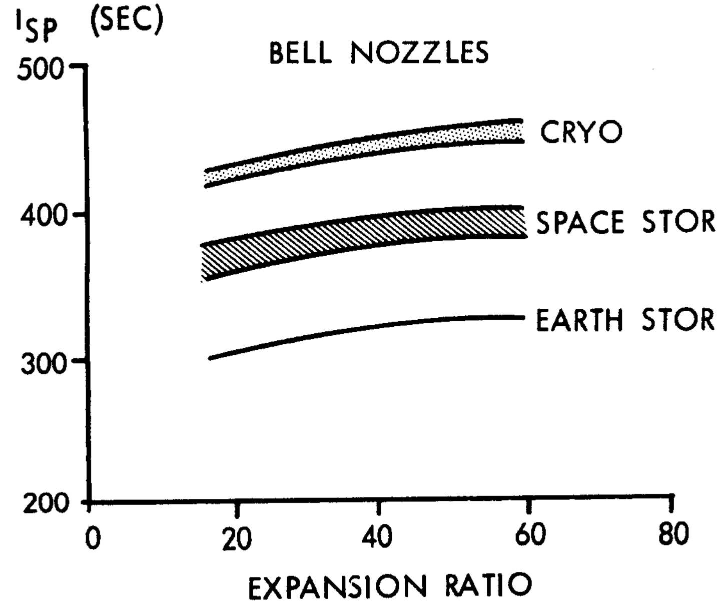
PROPULSION PARAMETERS

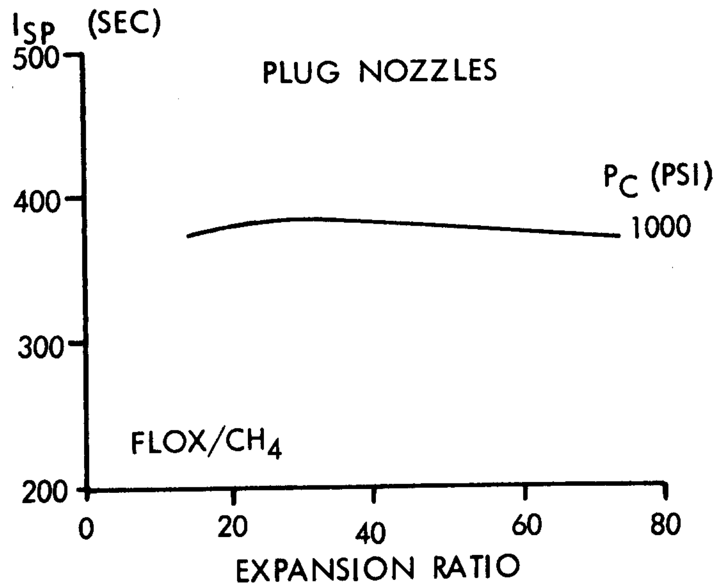
The maximum achievable ascent ΔV's are limited by propellant performance and volumes available for propellant storage. Minimum MEM weights are obtained with the cryogenics (as a consequence of their high specific impulse) followed in turn by $FLOX/CH_4$, OF_2/MMH, and the Earth storables. However, if a configuration size constraint is imposed (i.e., 31.5 feet), the ascent ΔV is limited to about 19,700 fps (6 km/sec) with fluorine/hydrogen and even the minimum requirement of 16,000 fps (4.9 km/sec) cannot be achieved with oxygen/hydrogen. Both the space and Earth storables can achieve the maximum ascent ΔV of 20,350 fps (6.2 km/sec).

Both bell nozzle (including extendable skirt) and plug nozzle engines were considered. The performance of bell nozzles increases with expansion ratio whereas the performance of plug nozzles is relatively constant beyond an expansion ratio of 20 to 30 at chamber pressures below 1000 psi (70 kg/m^2). Increasing the chamber pressure of plug nozzle engines induces a loss in performance because higher transpirant coolant flow rates are required. Although a performance advantage of 4 percent can be realized with bell nozzles (if adequate packaging volume is available), plug nozzle engines were recommended because of their compact size. A deliverable specific impulse of 383 seconds was determined for the selected propellant combination; combined transpiration and ablative cooling is required.

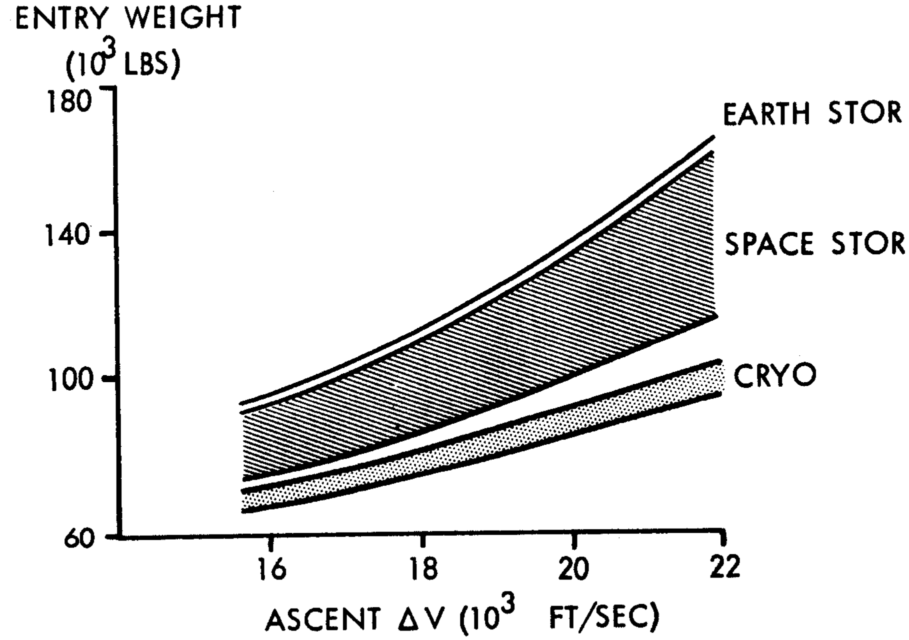
PROPULSION PARAMETERS

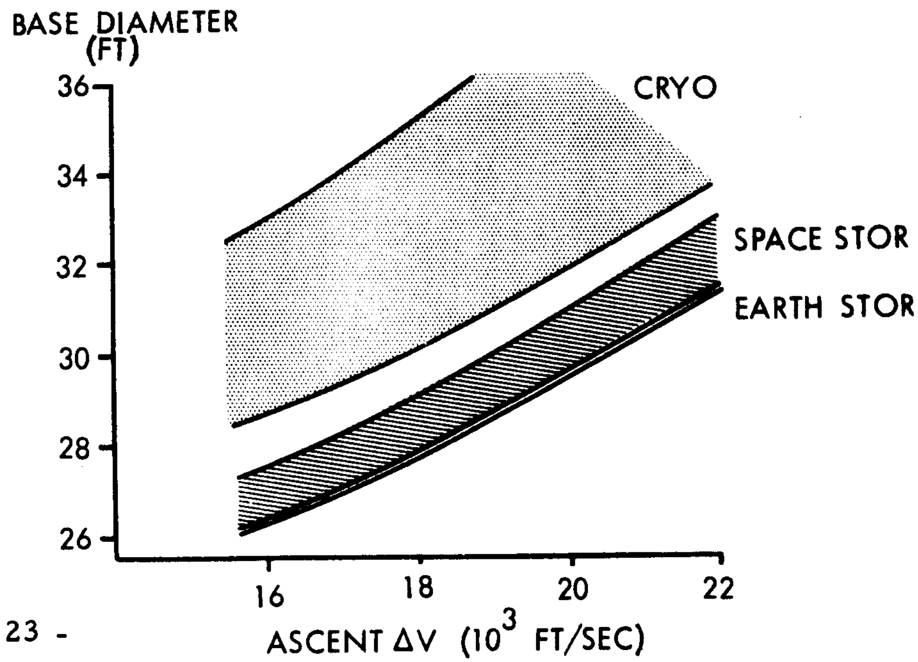
PERFORMANCE





PACKAGING





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PROPULSION SYSTEM

The recommended propulsion systems include an externally mounted solid propellant rocket motor which employs an advanced high energy propellant formulation for deorbit. The descent and ascent propulsion systems both employ $FLOX/CH_4$ and are constrained by the installation envelope. The propellant combination is hypergolic, although only the ascent stage requires restart capability. Both descent and ascent engines must be gimballed; the descent engine must be throttleable over a 10:1 ratio and the ascent engine must be restartable.

Reaction controls will be required during the orbital entry, ascent, and rendezvous phases. A typical high performance Earth storable propellant, i. e., CLF_5/MHF-5, has been selected for the RCS system.

PROPULSION SYSTEM

FUNCTION	SYSTEM
DEORBIT	SOLID MOTOR ADVANCED BERYLLIUM I_{SP} = 300 - 325 SEC VACUUM THRUST = 30-46K LB FOR 48 SEC P_C = 600 psi
DESCENT AND ASCENT	LIQUID ENGINES (PLUG NOZZLES, ϵ = 27) $FLOX/CH_4$ PROPELLANTS AT 5.75 MIXTURE RATIO P_C = 1000 psi (PUMP-FED) I_{SP} = 383 SEC VACUUM THRUST DESCENT = 100-140K LB - THROTTLEABLE 10:1 ASCENT = 30-35K LB - RESTARTABLE INERT HELIUM GAS PRESSURIZATION (20 - 30 PSIA)
RCS	LIQUID ENGINES (PULSING AND STEADY-STATE OPERATIONAL MODES) ClF_5/MHF-5 PROPELLANTS AT 2.40 MIXTURE RATIO P_C = 100 psi (PRESSURE-FED) INERT HELIUM GAS PRESSURIZATION - POSITIVE EXPULSION

RETARDATION AND LANDING

Retardation systems considered included a two-stage parachute system consisting of a hypersonic drogue and a subsonic main glide parachute with retrorockets; a single stage hypersonic drogue with retrorockets; and all retropropulsive descent system. Comparative weight fractions of the three concepts are shown for an m/C_LA of 6.4 slugs/ft^2 (1000 kg/m^2). The parachutes and ballutes were optimized to achieve minimum subsystem weight, assuming full redundancy of parachutes and ballutes.

The optimized ballute/retro system would employ a 60-foot (18-m) diameter ballute which is deployed at Mach 3.5 and decelerates the vehicle to approximately Mach 1.5. The ballute then is jettisoned and the retropropulsive maneuver is initiated. This is the lightest retardation system for the VM-7 atmosphere. It becomes relatively less attractive in more dense atmospheres or for lower m/C_LA's than 6.4 slugs/ft^2 (1000 kg/m^2). (Thinner atmospheres would require ballute deployment at higher altitudes and speeds.)

The ballute/retro system requires development of both the ballutes and the propulsion system. Although some small ballutes capable of Mach 3.5 deployment have been developed, the large sizes contemplated for the MEM represent a major technological development. Deployment of ballutes at appreciably higher velocities would introduce additional development issues (e.g., new materials to withstand heating). On the other hand, a retropropulsive system must be developed for either concept. An all retropropulsive descent system, although somewhat heavier, therefore is recommended because it combines the simplest system with the lowest development risk and costs.

RETARDATION AND LANDING

WEIGHTS

SYSTEM	PERCENT OF ENTRY WEIGHT		
	VM-7	VM-8	NASA UD&MD
2 STAGE CHUTE/RETRO	18-24	-	-
BALLUTE/RETRO	14-16	10-12	10-12
ALL RETRO			
L/D = 0.5	21	17	15
L/D = 1.0	24	18	17

CONCLUSIONS

60 FT , MACH 3.5 BALLUTE/RETRO SYSTEM LIGHTEST
BALLUTE WEIGHT ADVANTAGE DIMINSHED IN DENSER ATMOSPHERE

RECOMMENDATION

RETROPROPULSIVE DESCENT

SUBSYSTEMS (4-Man/30-Day MEM)

Various concepts for gas storage, CO_2 absorption, and water recovery were analyzed in selecting the 4-man/30-day MEM ECLSS subsystems. Cryogenic rather than high pressure storage of the consumables for the two gas atmosphere was selected because of weight and volume advantages. Oxygen recovery was not considered feasible because of the additional system complexity and weight, and short mission duration. A two-bed molecular sieve was selected for CO_2 absorption because of the weight, power, and volume advantages in comparison with a lithium hydroxide CO_2 removal system. To supplement the fuel cell water output, a multifiltration process for the recovery of wash water and condensates was selected on the basis of weight when compared to water storage and vacuum distillation/pyrolysis concepts. Total system weight is approximately 1540 pounds (700 kg).

The power requirements for the 4-man/30-day MEM are approximately 2 KW_e during descent and surface operations and 1.6 KW_e during ascent and rendezvous. Although an isotopic dynamic power source was considered for descent and surface operations, fuel cells (2 KW_e) were selected because of their advanced state of development and the by-product water produced. Two lunar module ascent batteries can provide power for the ascent and rendezvous phases. Total system weight is approximately 2740 pounds (1250 kg).

The integrated guidance and control system weighs approximately 190 pounds (86 kg). A high thrust RCS system is recommended during entry to provide the high roll rates for lift modulation, and pitch and yaw damping; low angular rates are recommended for the unpowered phases to minimize propellant consumption. Thrust vector control is used to provide attitude control during the powered descent and ascent phases (as in the lunar module); the roll engines provide roll control.

The MEM communications system links the MEM, the orbiting spacecraft, the Earth monitoring stations, and the crew on the surface. An S-band section provides line of sight MEM-spacecraft TV communications and MEM-Earth two-way voice, telemetry, tracking and ranging functions. The VHF link provides two-way voice communications between the MEM and the spacecraft and the MEM and crew on the surface. The rendezvous radar section consists of a pulse-type radar operating at L-band frequency with an interferometer in conjunction with a transponder located in the spacecraft. The system is similar to that used in Gemini. System weight is approximately 587 pounds (266 kg).

SUBSYSTEMS

SUBSYSTEMS 4-MAN/30-DAY	
ECLSS	GAS STORAGE - CRYOGENIC CO_2 REMOVAL - 2-BED MOLECULAR SIEVE H_2O MGMT - MULTIFILTRATION
EPS	DESCENT AND SURFACE OPERATIONS 2 KW_e FUEL CELL ASCENT 2 LM ASCENT BATTERIES
G & C	INTEGRATED G&N AND S&C FUNCTIONS (INERTIAL GUIDANCE, S&C ELECTRONICS, RENDEZVOUS AND LANDING RADAR, RADAR BEACON) ATTITUDE CONTROL UNPOWERED PHASE - ASCENT/DESCENT RCS POWERED PHASE - THRUST VECTOR CONTROL AND ROLL RCS
COMMUNICATIONS	SURFACE OPERATIONS S-BAND (MEM/SPACECRAFT TV, MEM/EARTH VOICE) VHF (MEM/SPACECRAFT VOICE, MEM/EVA VOICE, DATA) ASCENT AND RENDEZVOUS L-BAND (MEM/SPACECRAFT RADAR)

TEST PHILOSOPHY

The MEM test program philosophy is essentially the same as that used by the NASA on earlier manned spacecraft programs. The basic precepts of this philosophy are:

1. Tests will be initiated at the lowest level of hardware and will proceed, via the building block concept, to subsystem and, ultimately, to integrated systems tests.
2. Mission phase environments will be simulated, as closely as practicable, through Earth-based ground and flight testing to support qualification and verification.
3. The test program will reflect decoupling of major MEM subsystems and major spacecraft modules.
4. Crew safety issues will be identified early and resolved by means of exhaustive ground testing followed by unmanned flight testing.

TEST PHILOSOPHY

SAME AS FOR MERCURY, GEMINI & APOLLO

TEST AT LOWEST LEVEL

VERIFY PERFORMANCE FOR EACH MISSION PHASE

DE-COUPLE TESTING

MAJOR MEM SUBSYSTEM

MAJOR S/C MODULES

CREW SAFETY PARAMOUNT

TEST REQUIREMENTS

The various operational phases of the MEM, from acceptance of the vehicle through the docking at Mars just prior to MEM abandonment, are indicated. The column on the left lists the MEM subsystems, including structure and aerodynamic configuration. The matrix of open and solid circles indicates when the various subsystems are active; the dashed lines indicate the operational environment in which subsystems must survive in a dormant mode before becoming active. The typical mission timeline indicates the durations of the respective phases.

The solid circles identify the major development issues which require either an advance in the state-of-the-art (such as the descent and ascent propulsion subsystem) or involve a new application of current technology (such as use of the landing gear on Mars terrain).

The test requirements indicate that each subsystem should be tested under the conditions and environments which would exist when the subsystem is active (shown by the circles) and after the subsystem has been dormant for the appropriate length under environments similar to those shown by the dashed lines. For example, the descent reaction control system (RCS) must be acceptance tested; exposed to atmospheric conditions representing pre-launch operations for 30 days; boost levels of vibrations and loads; Earth orbital, space and Mars orbit vacuum and temperature conditions for 242 days; and then activated and tested under entry loads and temperatures. Simulation may be used where necessary to duplicate the loads, temperatures, and other required environments.

TEST REQUIREMENTS

OPERATIONAL PHASE

SUBSYSTEM	ACCEPTANCE	PRELAUNCH	BOOST	E.O. OPERATIONS	MARS INJECTION	TRANSIT	MARS CAPTURE	ORBITAL CHECKOUT	SEPARATION	DEORBIT	ENTRY	TERMINAL DESCENT	LANDING	MARS SURF. OPERATIONS	PRELAUNCH CHECKOUT	ASCENT	RENDEZVOUS	DOCKING
RCS - DESCENT	O	---	---	---	---	---	---	---	---	---	●							
ORBITAL	O	---	---	---	---	---	---	---	●	O	---	---	---	---	---	---	●	O
PROPULSION - RETRO	---	---	---	---	---	---	---	---	---	●								
- DESCENT	O	---	---	---	---	---	---	---	---	---	---	●						
- ASCENT	O	---	---	---	---	---	---	---	---	---	---	---	---	---	---	●		
GUIDANCE AND CONTROL	O	---	---	---	---	---	---	O	O	O	●	O	●	---	O	O	O	O
THERMAL PROTECT - H/S	---	---	---	---	---	---	---	---	---	---	●							
- INSUL	---	O	O	O	O	●	O	O	O	O	●	O	O	●	O	O	O	O
POWER SYSTEM	O	---	---	---	---	---	---	O	O	O	O	O	O	●	O	O	O	O
ECLSS	O	---	---	---	---	---	---	O	O	O	O	O	O	●	O	O	O	O
COMMUNICATIONS	O	---	---	---	---	---	---	O	O	O	●	O	O	O	O	O	O	O
BALLUTES/CHUTES	---	---	---	---	---	---	---	---	---	---	---	●						
MECHANISMS - DOCKING	---	---	---	---	---	---	---	---	---	---	---	---	---	---	---	---	---	O
- LANDING GEAR	---	---	---	---	---	---	---	---	---	---	---	---	●					
- SEP DEVICE	---	---	---	---	---	---	---	---	O	●	---	●	●	---	---	●		
STRUCTURE	---	---	●	---	O	---	●	---	---	---	●	●	●	---	---	●	---	O
AERODYNAMIC CONFIG											●	●						
CHECKOUT SENSORS	O	---	---	O	---	---	---	O	---	---	---	---	---	---	O			
TYPICAL TIMELINE (DAYS)	-90	-30	0	40	40	240	240	241	242	242	242	242	242	271	272	272	272	272

- - - DORMANT
O ACTIVE
● MAJOR DEV'T ISSUES

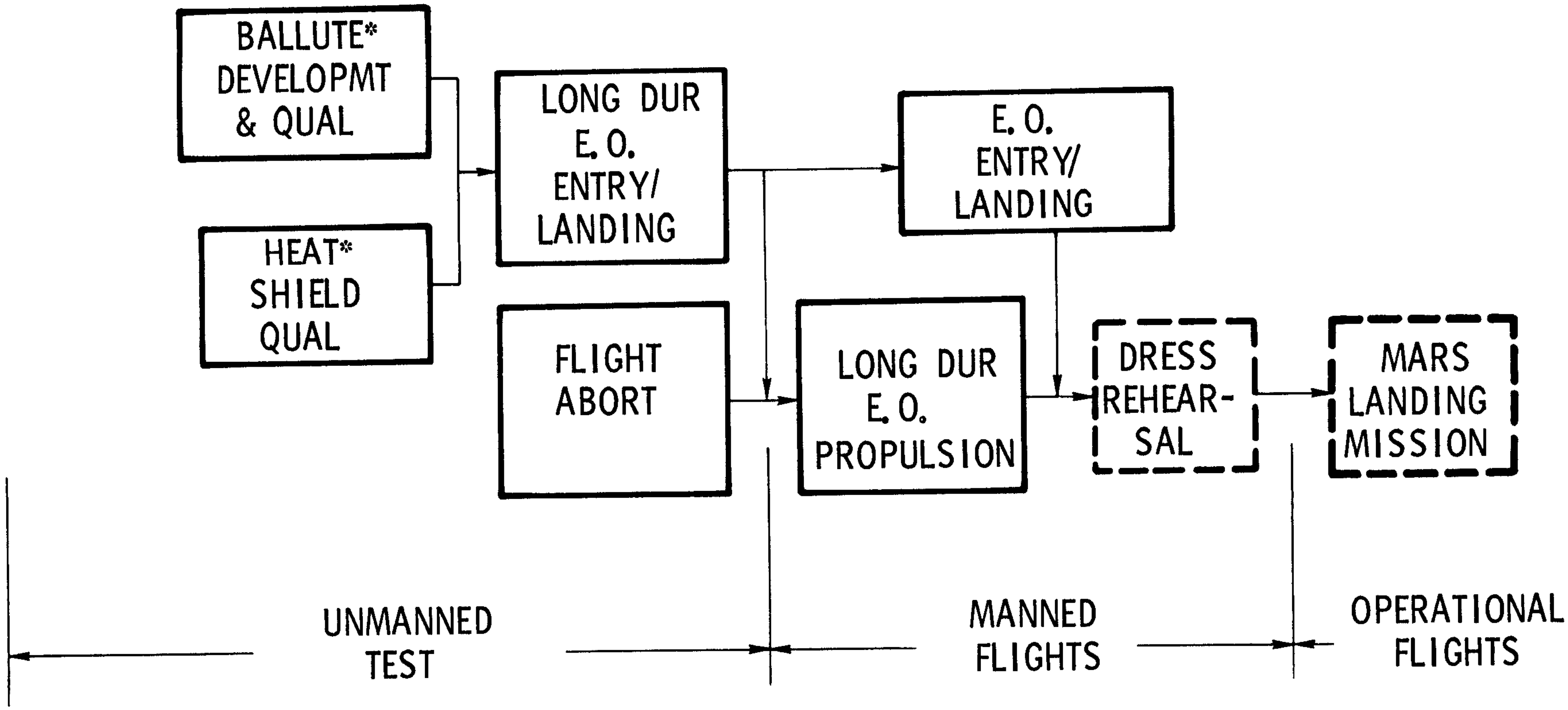
RECOMMENDED TEST PROGRAM

The recommended test program satisfies all test requirements at both the subsystems and integrated systems level with a minimum number of flight tests. Although retropropulsive descent and an Earth orbital entry heat shield design are recommended, the possibility of using ballutes and/or a Mars entry heat shield design is recognized.

The test program begins with ground testing at the component and subsystem level and builds up to ground testing at an integrated system level using "house" MEM's. The flight test program would start with development and qualification of the ballutes and qualification of the Mars heat shield, if these were incorporated in the MEM. Unmanned Earth entry and landing flight tests then would be conducted to demonstrate operation of the descent stage subsystems after extended times in Earth orbit (to simulate space soak during Mars transit) and to qualify the Earth orbital entry heat shield. This test would be followed by a manned flight to obtain entry and descent experience; extended stay times in orbit are not required. The other branch of the flight test program would start with an unmanned simulated abort test to verify the dynamics and aerodynamics of pre-landing abort as well as ascent engine operation. The next Earth orbital flight would be manned and test the descent and ascent propulsion subsystems, exercise of the rendezvous and docking maneuver, and the long-duration capability of the MEM would be demonstrated.

After completion of these tests, the MEM would be fully developed and qualified. A possible "dress rehearsal" flight of the integrated spacecraft, including the MEM in Earth orbit, might be considered before committing the Mars landing mission.

RECOMMENDED TEST PROGRAM



*IF BALLUTES & MARS ENTRY HEAT SHIELD ARE USED

TEST PROGRAM HARDWARE REQUIREMENTS

Eleven MEM test articles are required for the recommended test program including seven partial and four complete MEM's for the flight test program. The latter include two manned vehicles and a backup which could become the first deliverable MEM, if it is not required in the test program.

Three two-stage (S-IC/S-II) Saturn V's, two uprated Saturn I and three logistic vehicles are required to support the program. Although no major new facilities are required, existing environmental and propulsion testing facilities must be modified.

Should ballutes and a Mars entry heat shield be adopted, five additional partial MEM's (three for ballute and two for heat shield tests) would be required. In addition, a new solid booster of the Little Joe II type with a 33-foot (10m) diameter (for ballutes) and two launch vehicles with a diameter and capability equal to that of a first stage Saturn V (for heat shields) will be needed.

TEST PROGRAM HARDWARE REQUIREMENTS

RECOMMENDED

11 MEM TEST VEHICLES

7 PARTIAL	4 COMPLETE
IMPACT PROPULSION STATIC DYNAMIC ENVIRONMENTAL HOUSE FLIGHT ABORT	E.O. ENTRY/LANDING (UNMANNED) E.O. PROPULSION (MANNED) E.O. ENTRY/LANDING (MANNED) BACK-UP MEM

LAUNCH & SUPPORT VEHICLES

3 TWO-STAGE (S-IC/S-II) SATURN V'S 2 UPRATED SATURN -1'S 3 LOGISTIC SUPPORT VEHICLES	

BALLUTE & HEAT SHIELD*

5 ADDITIONAL PARTIAL MEM'S
12 NEW SOLID, LITTLE JOE II TYPE BOOSTERS
2 SINGLE STAGE (S-IC) SATURN V OR EQUIVALENT

*IF REQUIRED

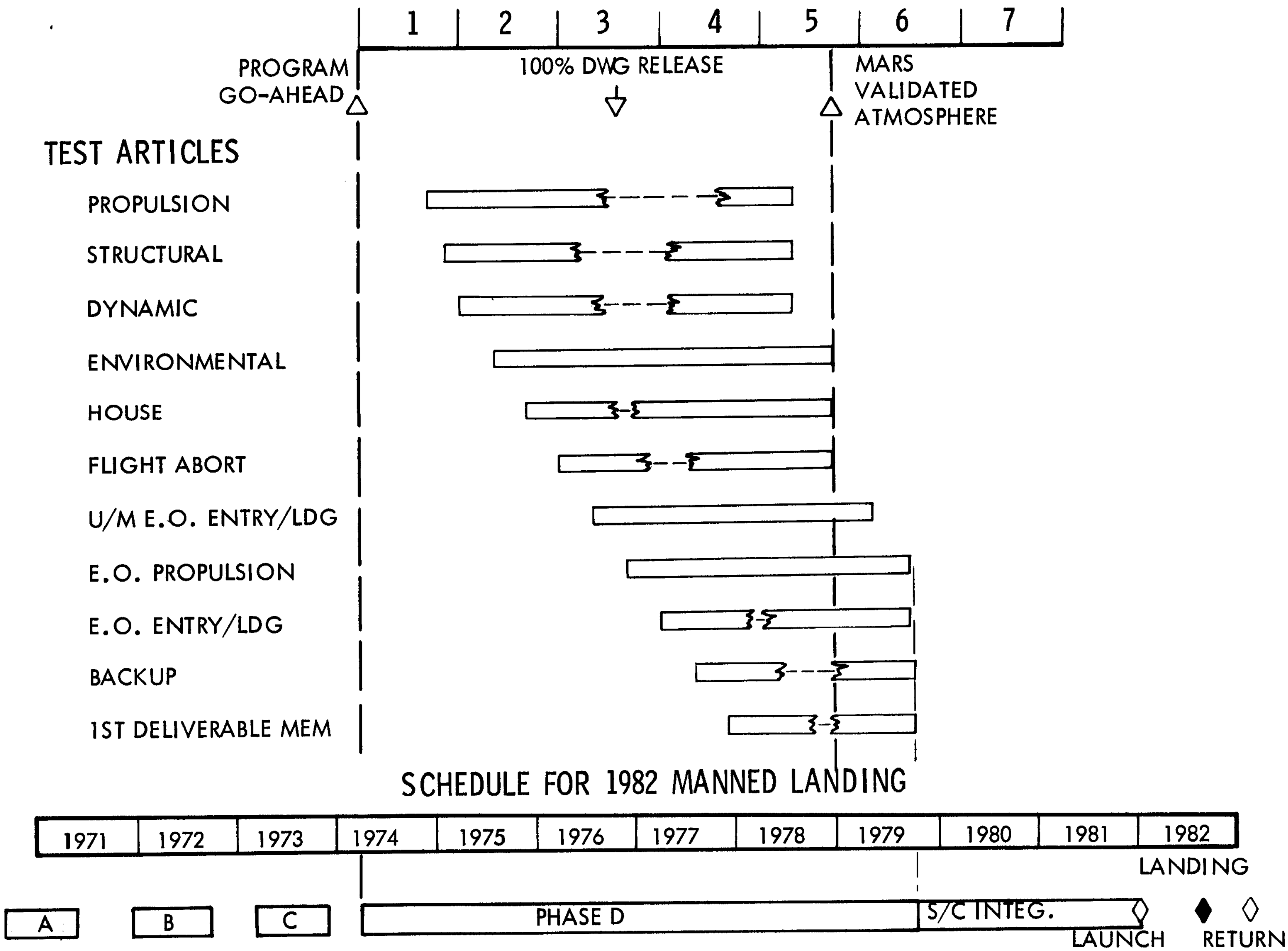
SCHEDULE SUMMARY

The schedule for the recommended MEM development and test program is presented based upon the assumptions of a maximum production rate of six vehicles per year, a progression from the simplest to the most complex vehicles, and manufacturing and system installation scheduling similar to the Apollo CSM. The program is success-oriented (i.e., assumes that all major tests will be successful) and does not include ballute or Mars heat shield tests.

Five and one-half years is required from Phase D program go-ahead to the delivery of the first qualified MEM to the spacecraft integration contractor. The assumption that a validated Mars atmospheric model will be available within four and three-quarter years from go-ahead is implicit in this schedule.

A Phase D go-ahead is required by the beginning of 1974 to achieve a 1981 Earth departure date for the 1982 manned Mars landing opportunity (the earliest opportunity considered). A two and one-half year period for delivery of the first MEM to the spacecraft integrator to the Mars mission launch, including a possible rehearsal flight, was assumed. This schedule is predicated Phase A and Phase B contracts in 1970 and 1972, and is compatible with the availability of a validated Mars atmosphere by as late as 1978.

SCHEDULE SUMMARY



MEM DEVELOPMENT COSTS

The estimated costs of the MEM development test program are compared for several MEM configurations and missions. Hardware costs include design, development, manufacture, and testing through qualification, but exclude those vehicles used in MEM spacecraft integration tests and/or in Mars mission operations tests. The total costs include the launch vehicles and associated operations, supporting logistics vehicles, new and modified facilities, and program support management.

For the recommended low L/D, 4-man/30-day MEM configuration designed for the elliptical orbit mission, the hardware development costs are $3.1 billion and the total program costs are $4.1 billion. If the weight increases 50 percent during the course of the program (as has been the experience on other spacecraft procurements), these costs may be expected to increase to $3.9 and $5.0 billion, respectively.

A 4-man/30-day MEM designed for the low circular orbit mission and without allowance for weight growth will cost a total of $3.7 billion. Sensitivities to other parameters for the low circular orbit mission can be summarized as follows:

- Mars heat shield tests increase total cost by $870 million
- Bullute tests increase total cost by $63 million
- A lifting body MEM design increases total cost by $240 million
- The low L/D configuration designed for 2-man/4-day low circular orbit mission will decrease the estimated total costs by $240 million.

MEM DEVELOPMENT COSTS

(IN $ BILLIONS)

COST ITEM	LOW CIRCULAR ORBIT	ELLIPTICAL ORBIT	
	NO GROWTH	NO GROWTH	50% WEIGHT GROWTH
MEM HARDWARE DEVELOPMENT	2.7	3.1	3.9
TOTAL COST	3.7	4.1	5.0

BASIS : 5-1/2 YEAR PROGRAM
4-MAN/30-DAY LOW L/D MEM
EARTH ENTRY HEAT SHIELD
RETROPROPULSIVE RETARDATION

COST SENSITIVITIES

COST ITEM	2-MAN/4-DAY	BALLUTE/RETRO RETARDATION	MARS ENTRY HEAT SHIELD	LIFTING BODY
MEM HARDWARE DEVELOPMENT	-0.22	+0.58	+0.76	+0.21
TOTAL COST	-0.24	+0.63	+0.87	+0.24

MAJOR TECHNOLOGY IMPLICATIONS

Technology implications which have been identified include:

1. A higher level of confidence in the Mars atmosphere, wind, and topography models than is currently available is desirable. Validation of the atmosphere, preferably by a Mars entry and soft landing probe prior to completion of the MEM qualification program is most desirable.

2. Compact high performance engines in the 25,000 to 140,000 pounds (11,000 to 64,000 kg) thrust range must be developed, together with the associated technologies for producing, handling, and testing the selected propellants. The selected propulsion system must consider requirements for other uses in the same time period. The recommended propulsion system employs plug nozzles at 1000 psi (70 kg/cm^2) chamber pressures and a combines transpiration and ablative cooling. $FLOX/CH_4$ are the recommended propellants.

3. New insulation concepts and/or materials are required which will protect propellants and equipment during the long transit times in space and on the Mars surface (where the atmospheric pressure nullifies super-insulation effectiveness). A concept suggested in this investigation utilizes insulation material encased in plastic bags which are evacuated and sealed in space. If hydrogen is used (e.g., for fuel cells) particular attention must be given to thermally isolating it.

4. If aerodynamic decelerators are included in the deceleration subsystem, a major technological development effort is required to accommodate 50,000 to 100,000 pound (23,000 to 45,000 kg) payloads at Mach 3 to 4 deployment speeds. A suggested sequence of testing would start with small-scale wind tunnel tests to develop materials and aerodynamic data in a simulated Mars atmosphere. This would be followed by the development of an experimental 1/3 to 1/2 scale system, including high speed, high altitude tests using Little Joe I or II type of boosters. The development of the full scale decelerators could then begin with confidence.

MAJOR TECHNOLOGY IMPLICATIONS

MARS ENVIRONMENT	HIGHER LEVEL OF CONFIDENCE IN MARS ATMOSPHERES, WINDS, AND TOPOGRAPHY
ENGINES	COMPACT, HIGH PERFORMANCE 25,000 - 140,000 LB THRUST ENGINES AND ASSOCIATED FUEL TECHNOLOGIES PLUG NOZZLE ENGINES TRANSPIRATION/ABLATIVE COOLING SPACE STORABLE PROPELLANTS
INSULATION	CONCEPTS/MATERIALS SUITABLE FOR LONG DURATION IN SPACE AND ON MARS SURFACE
AERODYNAMIC DECELERATORS	FOR 50,000 - 100,000 LB PAYLOADS MACH 3 - 4 DEPLOYMENT WIND TUNNEL & MATERIALS TESTS 1/3 TO 1/2 SCALE FLIGHT TESTS FULL SCALE DEVELOPMENT TESTS

SUMMARY AND CONCLUSIONS

The feasibility of the MEM concept has been demonstrated. The recommended design is capable of accomplishing even the most demanding Mars landing mission postulated, including rendezvous with the spacecraft in the most highly elliptical ($\epsilon \sim 0.9$) parking orbits, even allowing for 50 percent weight growth in the conceptual designs. The estimated weights range from 55,000 pounds (25,000 kg) for a 2-man/4-day low circular orbit mission to 109,000 pounds (49,500 kg) for the 4-man/30-day elliptical orbit mission. The recommended configuration has an $L/D \simeq 0.5$ and is aerodynamically similar to the Apollo command module. This configuration was compared to a lifting body ($L/D \sim 1.0$) and was found to be lighter, have more growth potential, and to offer fewer potential development problems.

The proposed MEM development and qualification program will satisfy all the test requirements identified. All the mission phases including entry, landing, surface operations, and ascent can be simulated in Earth tests by appropriate scaling of the test conditions, where necessary. No requirement has been identified for an unmanned test flight to Mars as part of the development and qualification program, and such flights therefore are not recommended. The recommended test program consists of a series of ground tests and flight tests.

The resources plan allows for a Phase D (design and development) go-ahead as late as 1974 to accomplish the assumed 1982 manned Mars landing. The schedule is success oriented; two and one-half years are provided for MEM/spacecraft integration prior to launching the mission. Acquisition of a validated model of the Mars atmospheric model by 1978 is mandatory to support the qualification testing program. Costs of the MEM development test program up to, but not including the first MEM delivered to the spacecraft integration contractor, have been estimated at $4.1 billion for the recommended configuration designed for the 4-man/30-day elliptical orbit mission. Weight growth (i.e., 50 percent) would raise the costs at least to $5.0 billion, whereas selection of a less ambitious 2-man/4-day circular orbit mission would reduce the anticipated costs to $3.4 billion. Incorporation of ballutes for retardation and a heat shield designed for Mars entry will increase the development program costs by $0.63 and $0.87 billion.

SUMMARY AND CONCLUSIONS

MEM DESIGN FEASIBLE

- LOW CIRCULAR TO HIGHLY ELLIPTICAL ORBITS
- LOW L/D CONFIGURATION RECOMMENDED
- W_G = 109,000 LB

EARTH TEST PROGRAM ADEQUATE

- ALL REQUIREMENTS SATISFIED
- UNMANNED MARS FLIGHT NOT REQUIRED
- GROUND TESTS + 4 FLIGHT TESTS RECOMMENDED

RESOURCES REQUIREMENTS

- 1974 PHASE D GO-AHEAD FOR 1982 LANDING
- VALIDATED ATMOSPHERE REQUIRED BY 1978
- COSTS \$4.1 $\bar{B}$ TO \$5.0 $\bar{B}$

MEM DESIGN BRIEFING OUTLINE

The requirements analysis and MEM conceptual designs are presented. After a review of the study groundrules, the considerations leading to the selection of the parking orbit and MEM design missions are discussed.

The presentation of the performance requirements follows the same sequence as the MEM mission, i. e., deorbit, entry and landing, ascent, and rendezvous. Both the minimum and maximum mission requirements were considered.

The design synthesis will discuss the aerodynamic characteristics of the candidate configurations. Primary consideration was given to those subsystems which required critical design decisions, namely the propulsion, retardation and thermostructural systems, and the auxiliary subsystems. Weight considerations, sensitivities and tradeoffs will be discussed as a basis for arriving at the recommended MEM design.

MEM DESIGN
BRIEFING OUTLINE

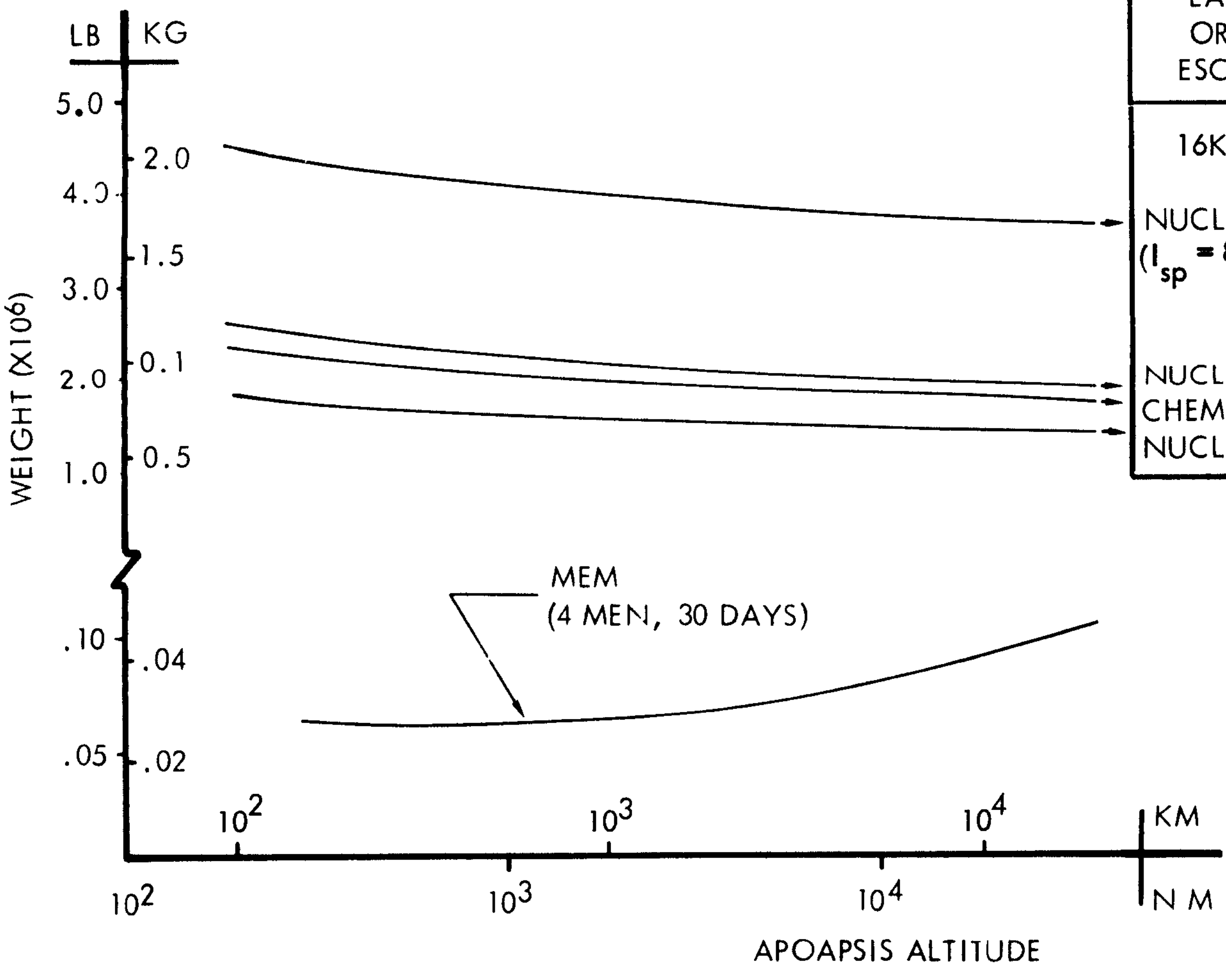
MISSION ANALYSIS	PARKING ORBIT TRADE-OFFS SCIENTIFIC OBJECTIVES SELECTED MISSIONS
PERFORMANCE REQUIREMENTS	DEORBIT ENTRY & LANDING ASCENT & RENDEZVOUS
DESIGN SYNTHESIS	AERODYNAMIC CONFIGURATIONS SUBSYSTEMS WEIGHTS
CONCLUSIONS & RECOMMENDATIONS	

INITIAL WEIGHT IN EARTH ORBIT

Typical variations of initial weight in Earth orbit of the spacecraft as a function of the apoapsis altitude of the parking orbit at Mars are shown for aerobraker and retrobraker transit vehicles. Both chemical and nuclear propulsion modes for trans-Mars injection were considered. These weights are based on typical ΔV requirements of 16,000 fps (4.9 km/sec) for Earth orbit escape and 15,000 fps (4.6 km/sec) for Mars retrobraking. Of the 10 mission opportunities between 1980 and 1999 the aerobraker spacecraft can accomplish 9 and the retrobraker 8 of the missions. The data were derived from a concurrent study on the Technology Requirements for Mars/Venus Aerobraking (NAS2-4135). A 25 percent reduction in the initial weight in Earth orbit can be achieved with highly eccentric orbits at the expense of increased MEM weight.

INITIAL WEIGHT IN EARTH ORBIT

PERIAPSIS ALTITUDE = 270 N MI (500 KM)
MEM I_{sp} = 383 SEC



EARTH ORBIT ESCAPE	MARS ORBIT CAPTURE	MARS ORBIT ESCAPE
16K FPS	15K FPS	16-10.5K FPS
NUCLEAR (I_{sp} = 800 sec)	CHEMICAL (I_{sp} = 400 sec)	CHEMICAL (I_{sp} = 400 sec)
NUCLEAR	NUCLEAR	NUCLEAR
CHEMICAL	AEROBRAKER	CHEMICAL
NUCLEAR	AEROBRAKER	CHEMICAL

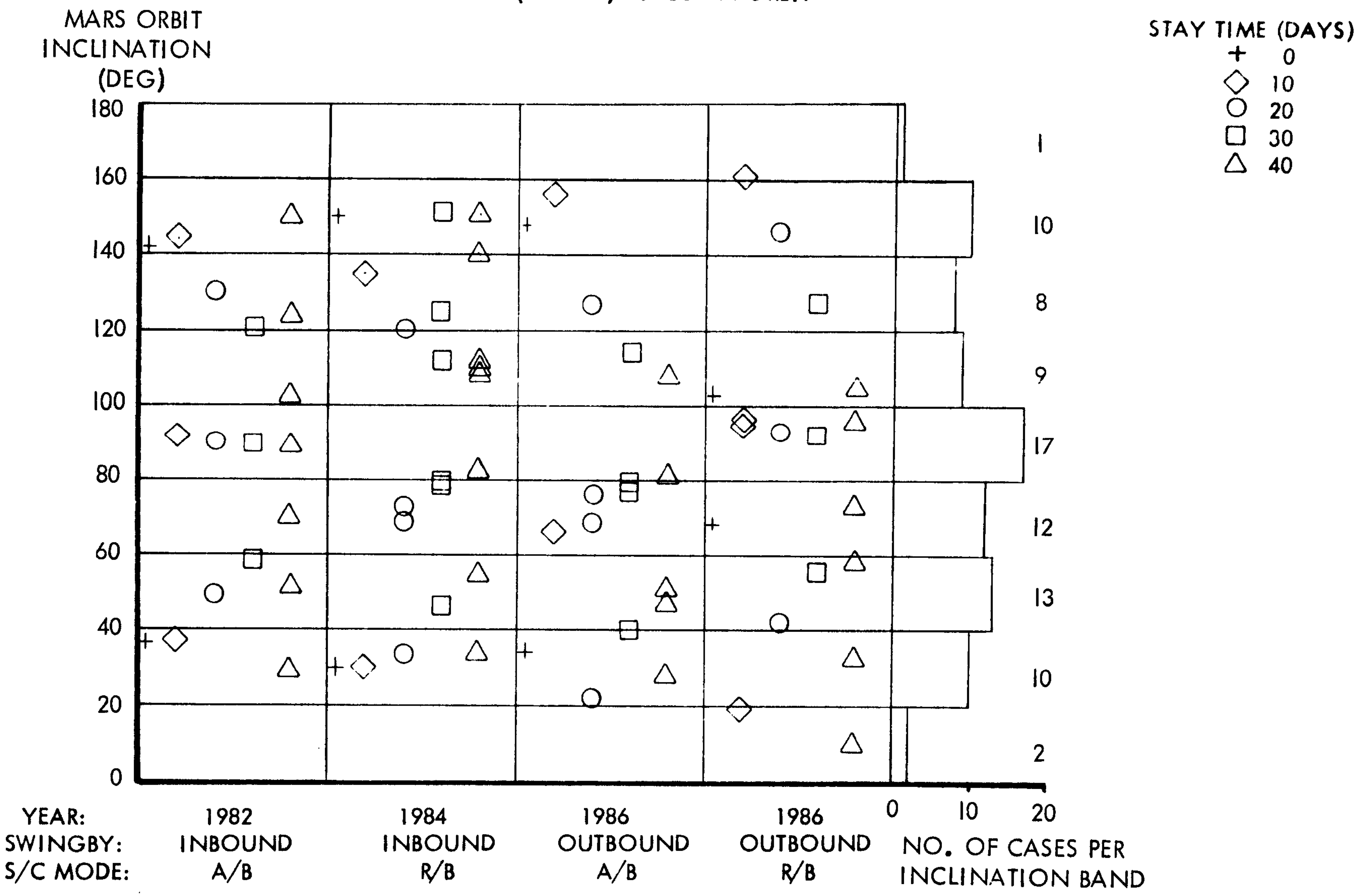
AVAILABLE PARKING ORBIT INCLINATIONS

VENUS SWINGBY MISSIONS

A study of the Venus swingby mode for a number of opportunities resulted in identifying compatible parking orbit inclinations. For these inclinations no out-of-plane orbital changes are needed at the end of the specified stay times at Mars. The chart summarizes the results for 1982, 1984, and 1986 missions with aerobraker or retrobraker spacecraft, for a 270 NM (500 KM) orbit about Mars. Statistically the orbits shown are distributed over a broad range of inclinations between 20 and 160 degrees.

AVAILABLE PARKING ORBIT INCLINATIONS
VENUS SWINGBY MISSIONS

270 NM (500 KM) CIRCULAR ORBIT



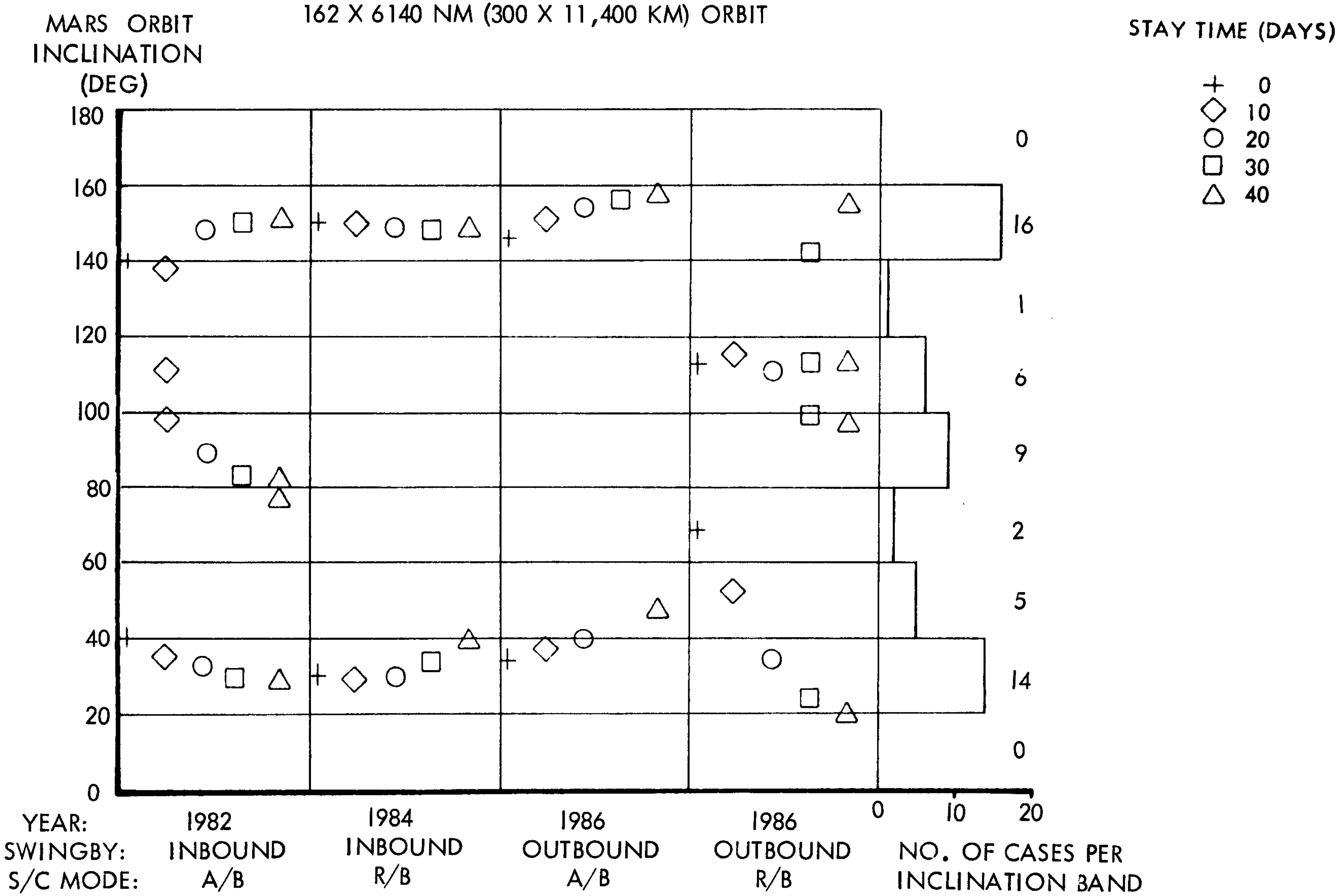
AVAILABLE PARKING ORBIT INCLINATIONS

VENUS SWINGBY MISSIONS

This chart indicates the compatible orbit inclinations for the same Venus swingby missions as shown in the last chart, but for an elliptical orbit about Mars of 0.6 eccentricity. The favored inclinations are grouped in two bands at inclinations of about 20° to 40° and 140° to 160°. If non-compatible inclinations are desired, an out-of-plane maneuver is required by either the MEM or the spacecraft; large plane changes, however, can be made for only a small delta V expenditure at the apoapsis of highly elliptical orbits.

AVAILABLE PARKING ORBIT INCLINATIONS VENUS SWINGBY MISSIONS

162 X 6140 NM (300 X 11,400 KM) ORBIT

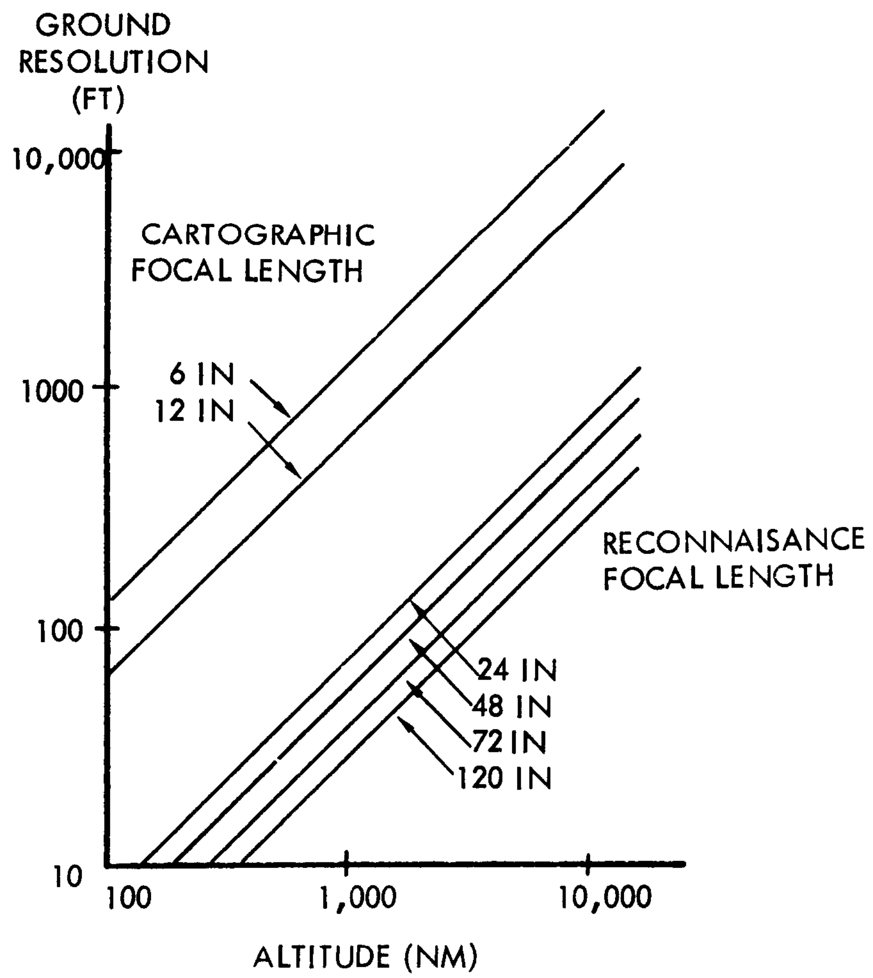


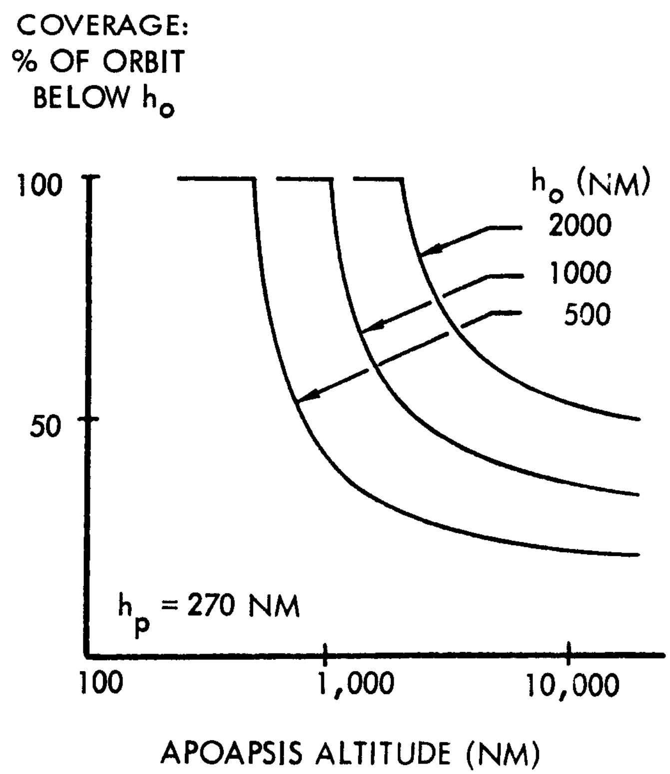
PHOTOGRAPHIC RESOLUTION AND COVERAGE

Photographic mapping of the planet is a prime consideration in selecting the spacecraft parking orbit. Ground resolutions on the order of 100 feet (~30 m) can be achieved with cartographic or 10 feet (~3 m) with sophisticated reconnaissance cameras from low circular orbits. High resolution photography is feasible on only a fraction of an elliptical orbit. For example, a resolution of 15 to 40 feet (4.6 to 12 m) can be obtained with a reconnaissance lens at an altitude of 500 nm (930 km); however, in a 270 by 6140 nm (500 by 11,400 km) orbit, only 25 percent of the trace is below 500 nm.

The planet can be mapped from orbits which have short periods in the 4- to 30-day mission durations considered. Orbits with apoapsis altitudes greater than 8500 nm (15,700 km) and periods of 8 hours or more seriously reduce resolution.

PHOTOGRAPHIC RESOLUTION AND COVERAGE





AREAS OF INTEREST ON MERCATOR PROJECTION

Examination of the scientific objectives as set forth by the Space Science Board of the National Academy of Science at Woods Hole indicated the following preferred areas for a manned landing:

a. Dark areas, including the wave of darkening and polar regions for biological experiments

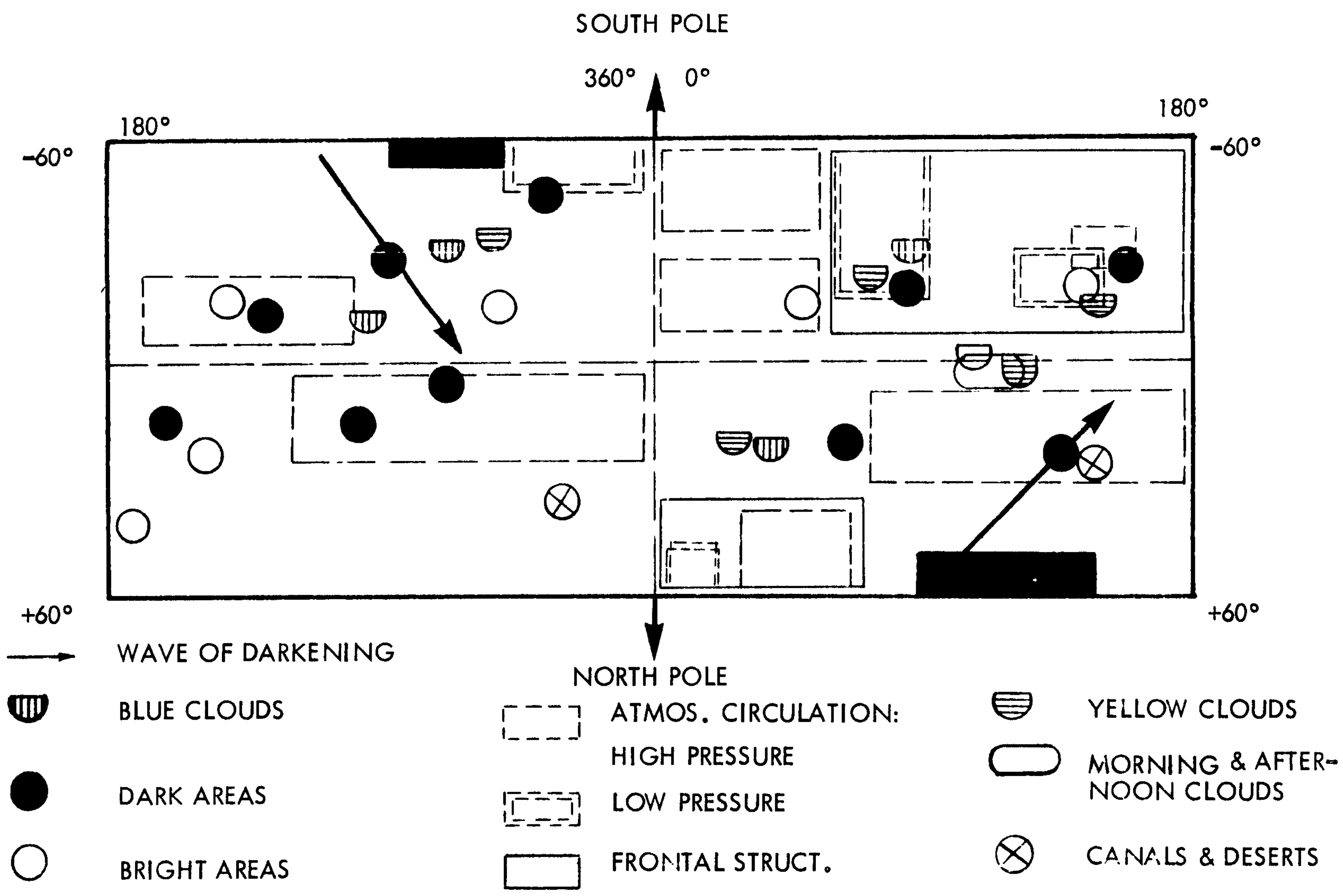
b. Blue and yellow cloud areas for atmospheric experiments

c. Dark and bright areas and the polar caps for areological and areophysical experiments

d. Equatorial regions for exosphere and magnetosphere experiments

This map shows schematically the location of possible areas of interest. Since detection of life is a primary scientific objective, the dark areas (shown in black) at approximately 60 degrees north and 60 degrees south latitude represent potential landing areas.

AREAS OF INTEREST ON MERCATOR PROJECTION



SELECTED MISSIONS

Considerations of the initial weight in Earth orbit, possible parking orbit inclinations and scientific objectives led to the selection of two MEM design missions. The first was a 4-man/30-day low circular orbit mission. An orbital altitude of 270 nm (500 km), inclination of 70 degrees, and landing site latitude of 63 degrees were selected. This mission was compatible with the orbital and scientific requirements and represented the simplest operational mission in terms of entry velocity and ascent ΔV capability.

The second selection was an elliptical orbit mission which was derived from a concurrent NASA/NAR contract on aerobraker spacecraft technology (NAS2-4135). The orbital parameters were 162 by 36,100 nm (300 x 66, 900 km) at an inclination of 90 degrees (to afford a landing capability anywhere on the planet). This mission is characterized by the maximum entry velocity and ascent ΔV capability requirements. Although this mission results in the heaviest MEM, the spacecraft weights and initial weights in Earth orbit are minimized because of the reduced spacecraft ΔV requirements.

SELECTED MISSIONS

$\boxed{\text{LOW CIRCULAR ORBIT MISSION}}$	4 MEN/30 DAYS 270 NM (500 KM) ORBIT MINIMUM ENTRY VELOCITY AND ASCENT ΔV 70° ORBIT INCLINATION 63° LANDING LATITUDE
$\boxed{\text{ELLIPTICAL ORBIT MISSION}}$	4 MEN/30 DAYS 162 X 36,100 NM (300 X 66,900 KM) ORBIT MAXIMUM ENTRY VELOCITY AND ASCENT ΔV 90° ORBIT INCLINATION

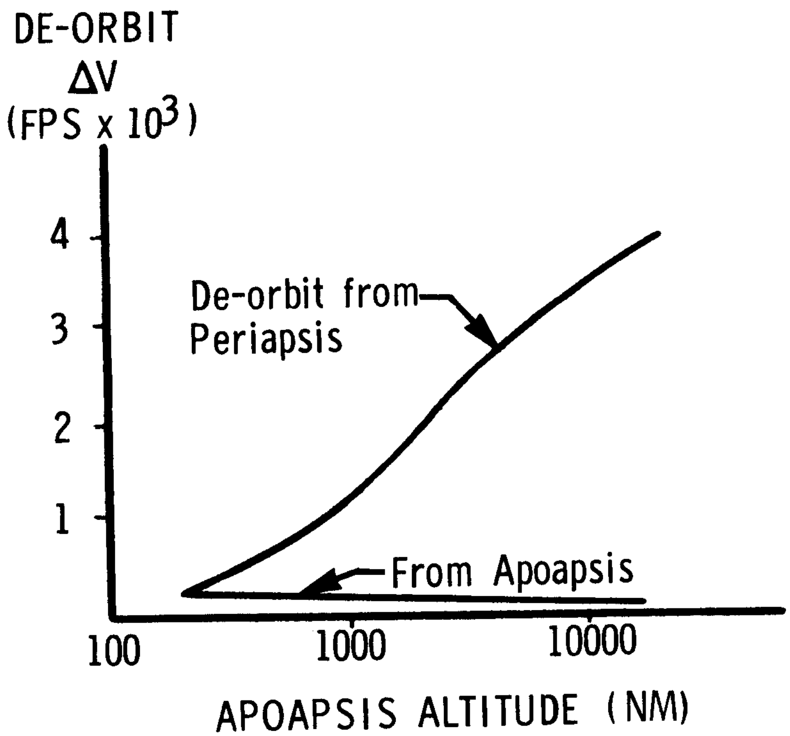
DEORBIT AND ENTRY

Deorbit ΔV requirements vary as a function of the periapsis and apoapsis altitudes and on the true anomaly at which the maneuver is carried out. The figure on the left illustrates the ΔV requirement for a periapsis altitude of 270 nm (500 km). This is less than 200 fps (61 m/sec) if injection occurs at apoapsis, but increases rapidly with apoapsis altitude if injection is at periapsis.

The figure on the right illustrates the entry corridor in terms of entry angle as a function of apoapsis altitude for periapsis altitude of 270 nm (500 km). The lower limit of the corridor is defined by skipout, while for angles above the upper boundary, pull-up occurs below the minimum acceptable altitude of ~33,000 feet (10 km). The illustration is for a value of m/C_LA of 10 slugs/ft^2 (1570 kg/m^2). For lower values of m/C_LA the entry corridor widens on the pull-up side.

A guidance error of less than ±0.5 degrees is projected and, therefore, no difficulty is expected in entering from orbit within the corridor shown.

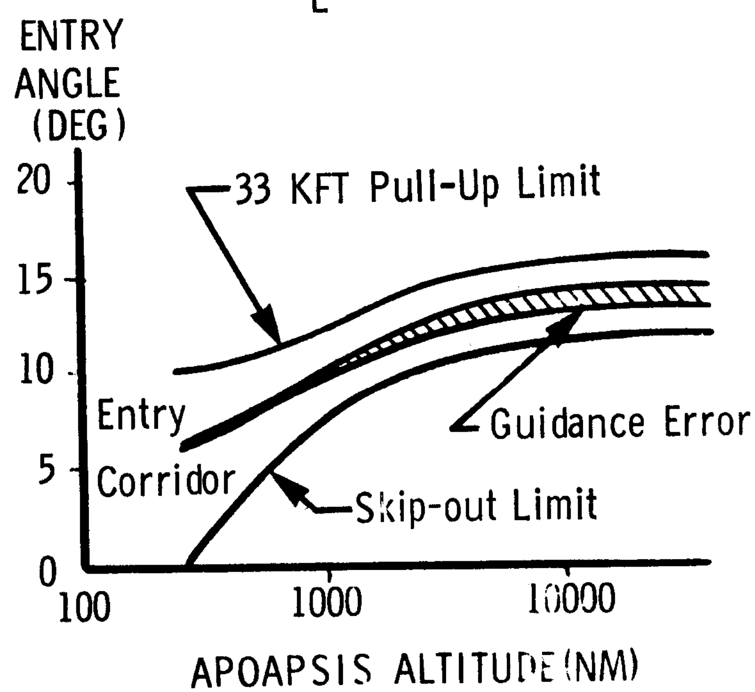
DEORBIT & ENTRY



h_p = 270(NM)

Deorbit from Apoapsis

$M/C_L A$ = 10 slugs/ft^2

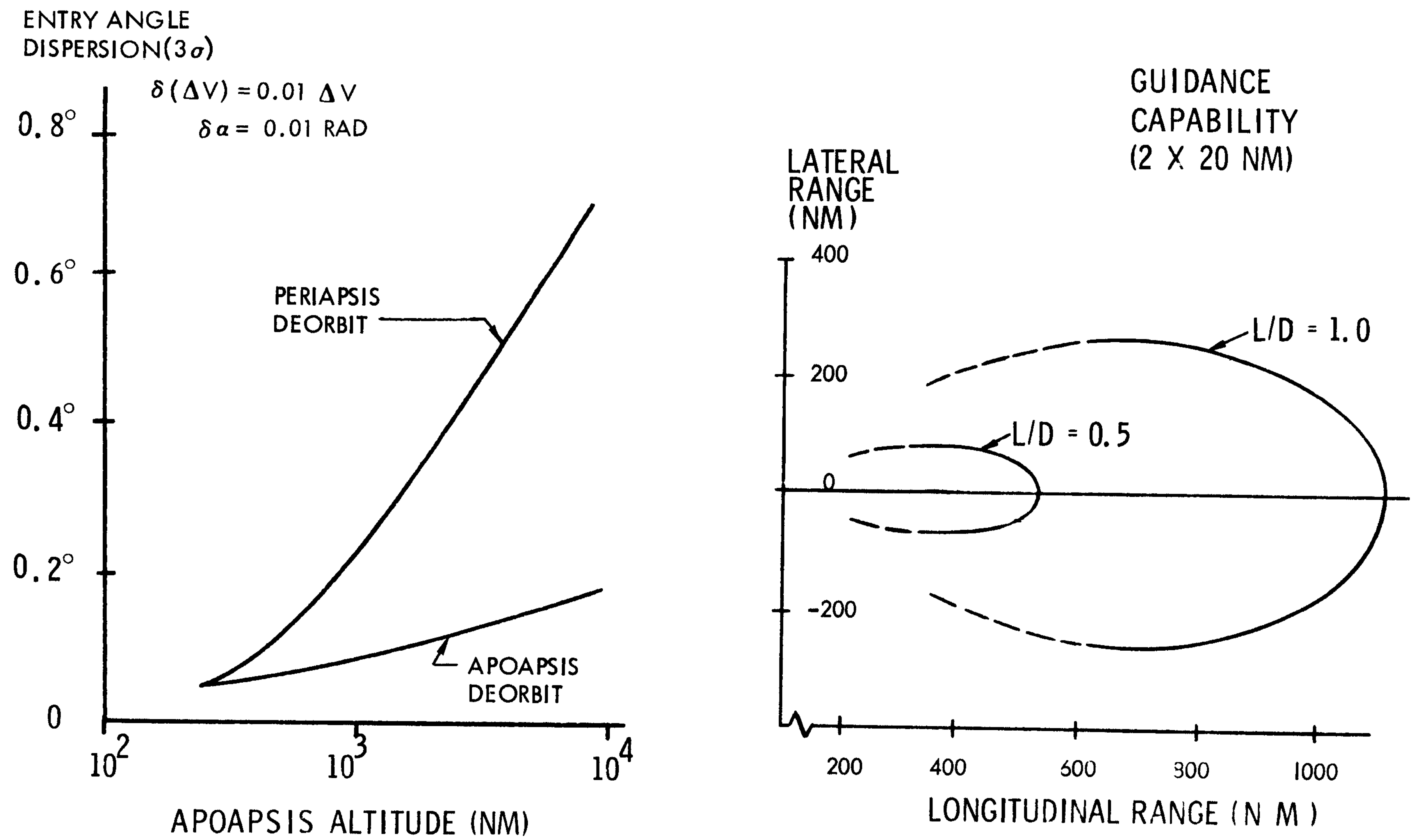


ENTRY GUIDANCE AND FOOT PRINT

Entry angle dispersions due to errors in initial orbit determination and deorbit were analyzed. Based on conservatively estimated errors of 1 percent in the deorbit ΔV and 0.01 radians in deorbit direction, 3σ entry angle dispersions can be held to less than 0.2 degree for deorbit from apoapsis. Dispersion, however, increases if deorbit occurs at periapsis because of the considerably larger ΔV required to deorbit. Deorbit at apoapsis or near apoapsis is therefore preferred inasmuch as less ΔV is required and more accuracy is possible.

Estimated ranging capabilities, based on an equilibrium glide trajectory, are shown on the right. The landing footprint for a vehicle L/D of 0.5 is approximately 120 by 500 nm (220 by 920 km); an L/D of 1.0 will yield a 500 by 1100 nm (920 by 2000 km) footprint. The point at which entry begins can be controlled by deorbiting at the appropriate point in the orbit; additional lateral range can be achieved by adding modest lateral components in the deorbit ΔV. The guidance capability following entry is estimated to be on the order of 2 by 20 nm (4 by 30 km). Since requirements for a minimum ranging capability were not identified, either and L/D of 0.5 or 1.0 is considered adequate for the MEM.

ENTRY GUIDANCE AND FOOTPRINT

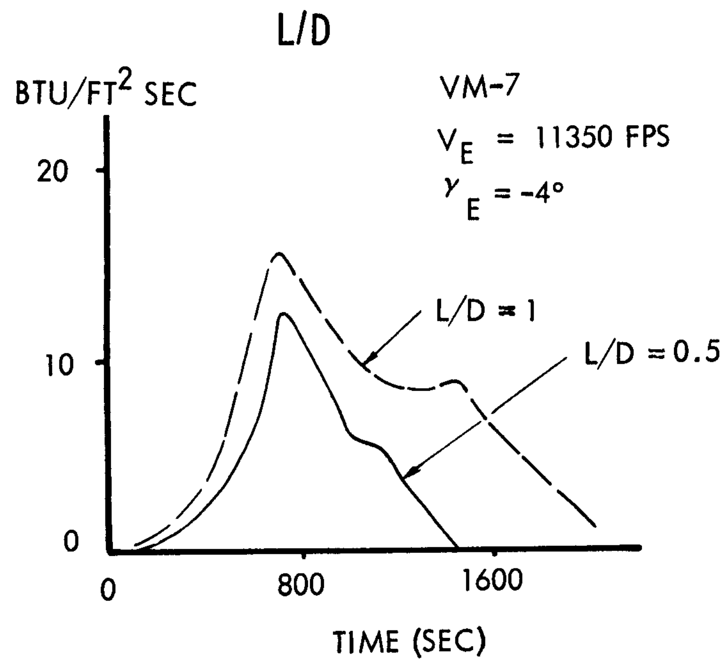


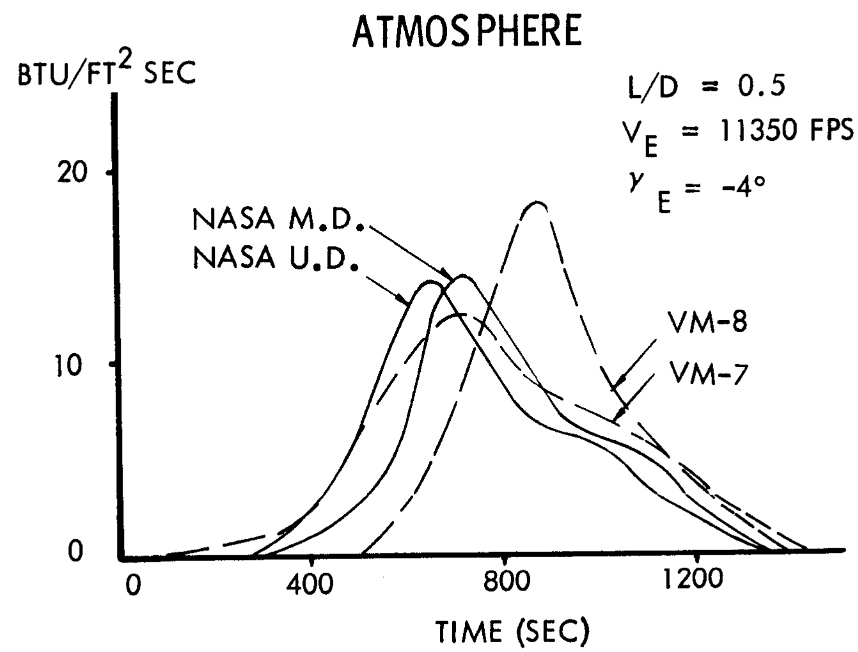
ENTRY HEATING SENSITIVITIES

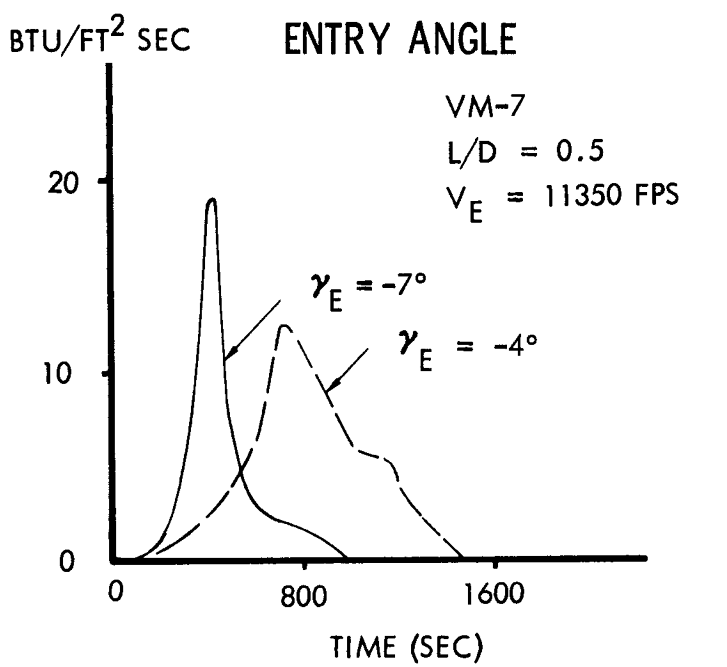
Typical variations of the entry heating pulse are illustrated. The several cases refer to the stagnation point heating rates of typical low L/D and lifting body MEM's with $m/C_L A$ values of 6.4 slugs/ft^2 (1000 kg/per m^2). The heating pulses last 1000 to 2000 seconds. Heating rates and total heat loads tend to be greater for the lifting body (L/D = 1) than for the low L/D configuration (L/D = 0.5). The peak heating rates are greatest in the VM-8 atmosphere and least in VM-7, while the total heat loads are greatest in VM-7 and least in VM-8. Intermediate heating rates and loads are encountered in the NASA upper and mean density model atmospheres. These data are correlated to the scale height, which is considerably larger in VM-7 than in VM-8, and intermediate in the NASA model atmospheres. Increasing the entry angle from -4 to -7 degrees increases the peak heating rates by 80 percent, but decreases the total heat load.

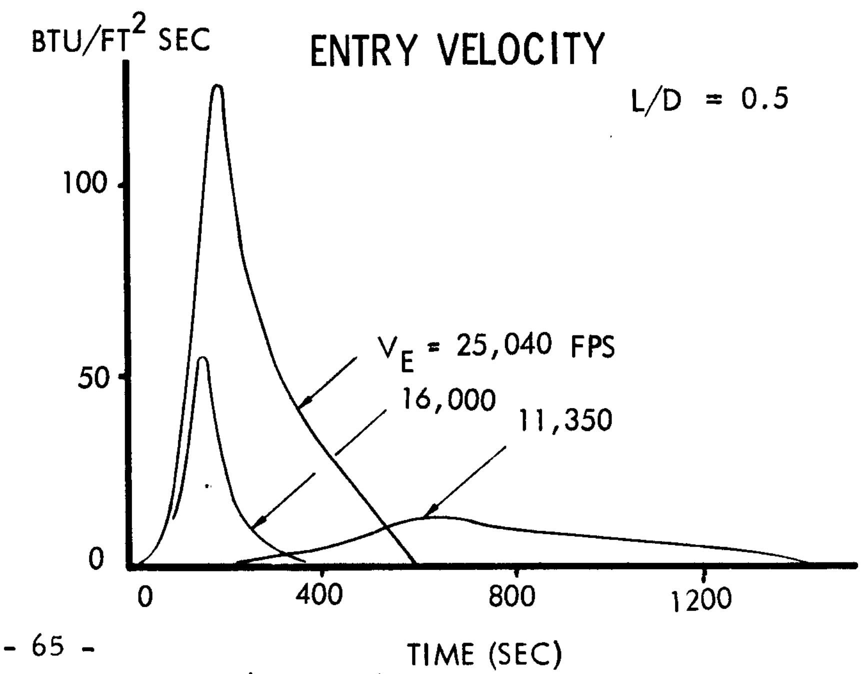
Mars entry from low circular and high elliptical orbits are compared to Earth entry from a low circular orbit. The peak heating rates for the three trajectories are 12, 55, and 102 BTU/ft^2 sec (14, 62, and 115 watts/cm^2). The heat loads for the two Mars entry trajectories are similar (i.e. 6800 BTU/ft^2 (7700 joules/cm^2); the steeper entry angle required for entry from the elliptical orbit compensates for the effect of the higher entry velocities. The heat load for the Earth entry, however, is nearly three times greater.

ENTRY HEATING SENSITIVITIES









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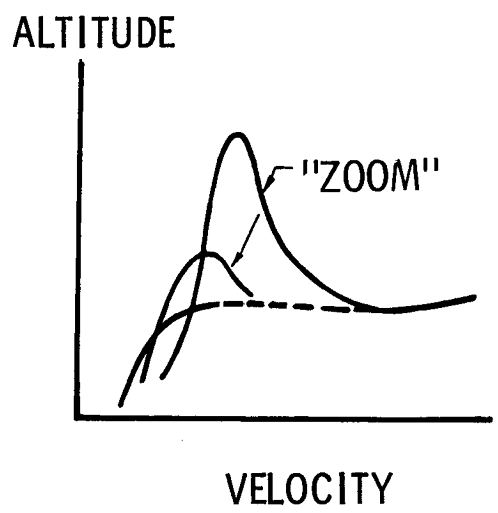
TERMINAL VELOCITIES

Following entry into the Mars atmosphere, the MEM decelerates to an altitude and velocity from which the final retardation sequence (e. g. ballutes and/or retropropulsive descent) is initiated. A number of possible flight modes are shown schematically in the altitude-velocity diagram on the left. A horizontal flight path at some predetermined altitude may be maintained by modulating the lift vector, as shown by the dotted line, until equilibrium conditions are reached (i. e., when lift, centrifugal force, and weight are balanced). Full positive lift may be applied before the velocity has decreased to the equilibrium value, in which case a "zoom" maneuver results and the vehicle climbs to a higher altitude. Depending on the velocity at which the zoom maneuver is initiated different velocity-altitude combinations can be achieved at the summit of the zoom. The velocity-altitude envelope is shown in the figure on the right for an $m/C_L A$ value of 6.4 slugs/ft^2 (1000 kg/m^2), and represents the lowest velocity/maximum altitude combinations which can be achieved. At low altitudes, the VM-7 presents the worst design case of the four atmospheres considered, since the terminal velocity is largest.

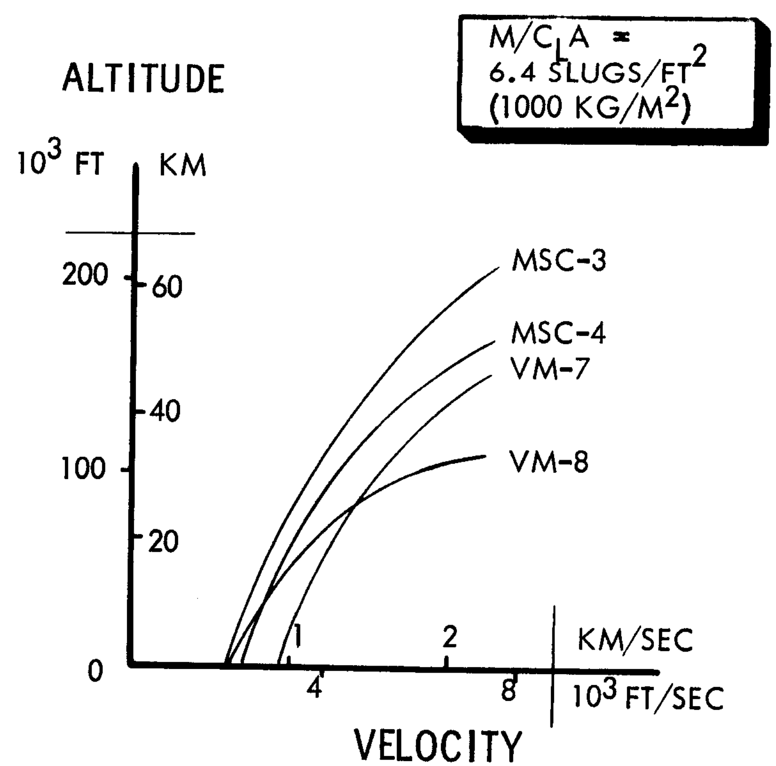
The zoom maneuver is advantageous for the ballute or parachute-retropropulsion modes, wherein the maximum possible altitude is desirable for deployment. For the retropropulsive descent mode, however, altitude is not a problem and it is acceptable to maintain level flight down to equilibrium conditions.

TERMINAL VELOCITIES

ZOOM MANEUVER



ENVELOPE OF ZOOM MANEUVERS

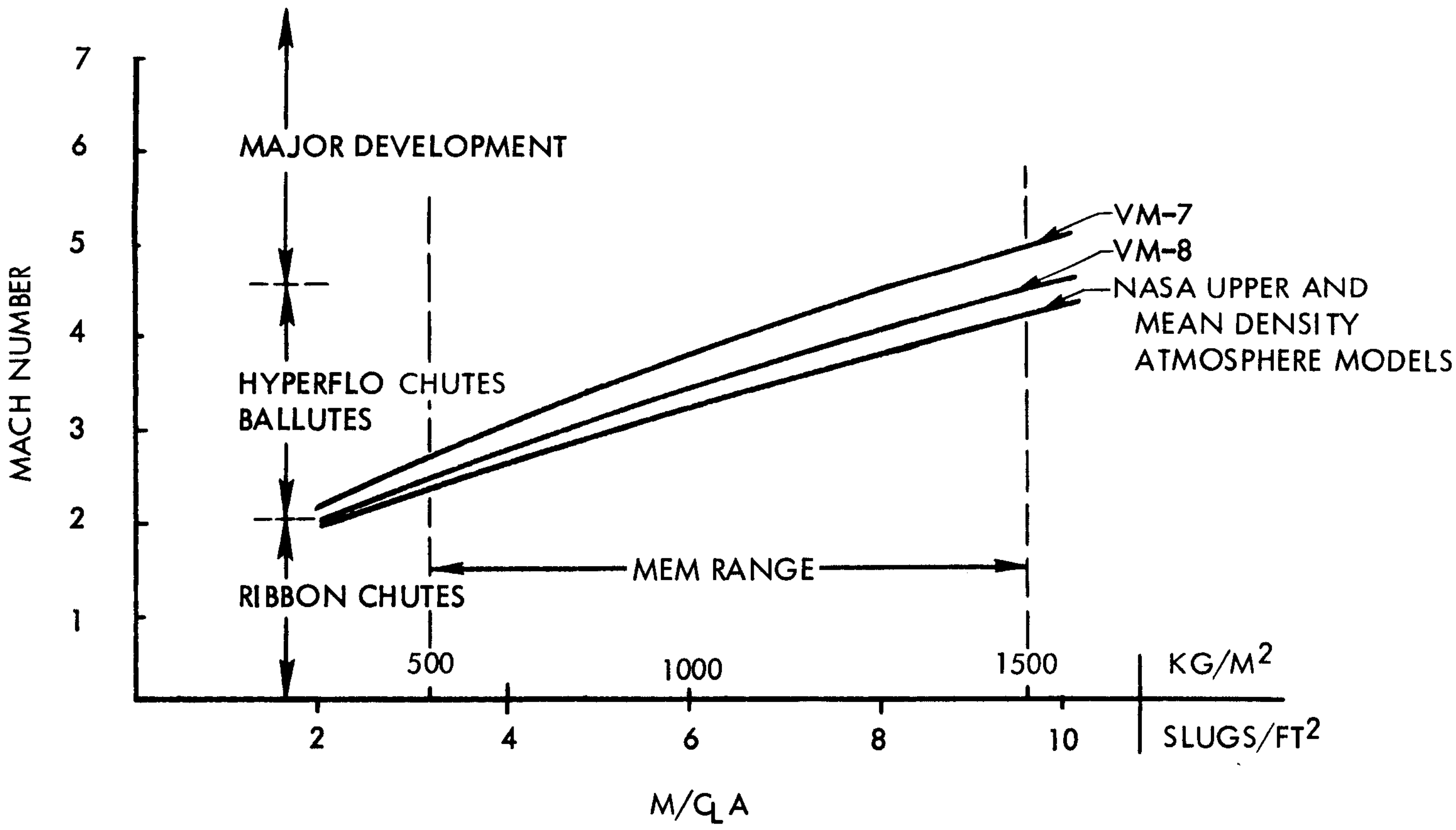


DEPLOYMENT MACH NUMBER FOR AERODYNAMIC DECELERATORS

For aerodynamic decelerators, such as parachutes or ballutes, deployment Mach numbers which yield adequate altitude for deceleration and retropropulsion must be chosen. The selection of deployment Mach number depends on the atmospheric model considered and the m/C_LA of the vehicle. For the range of m/C_LA values of interest, deployment Mach numbers vary from 2.3 for the lowest m/C_LA in the NASA upper and mean density models to 4.8 for the highest m/C_LA in VM-7.

The deployment Mach number is important because it determines the type of decelerators that can be used. For the MEM the choice among existing devices is limited to ballutes and hyperflo parachutes. Ribbon parachutes are ruled out as initial drogue devices, but may be used as second-stage parachutes at lower Mach numbers. For the highest values of m/C_LA, and particularly in the VM-7 atmosphere, major technological developments may be required because of both stability and heating problems.

DEPLOYMENT MACH NUMBER FOR AERODYNAMIC DECELERATORS

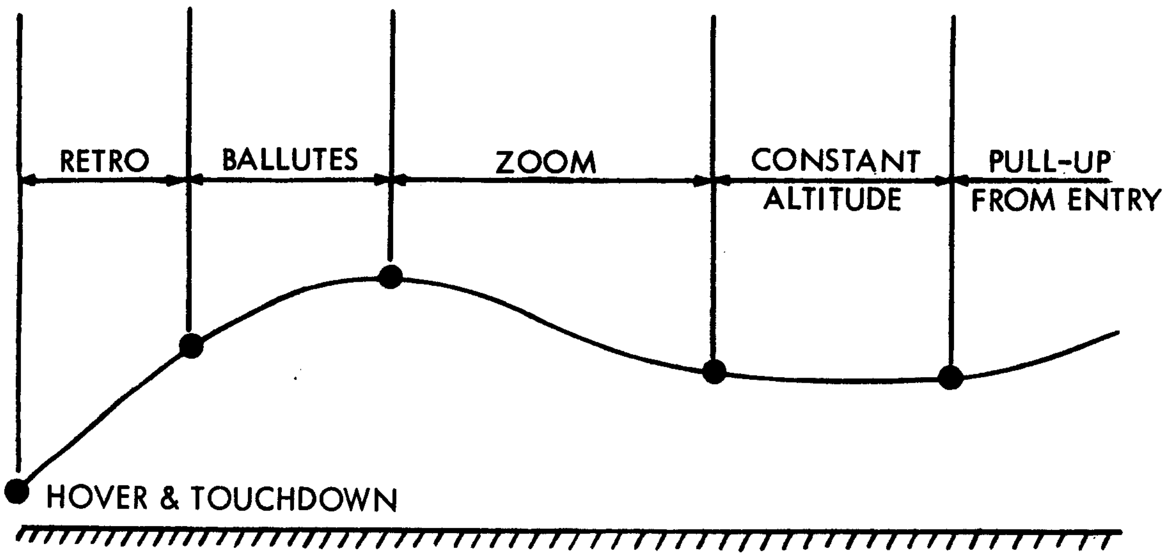


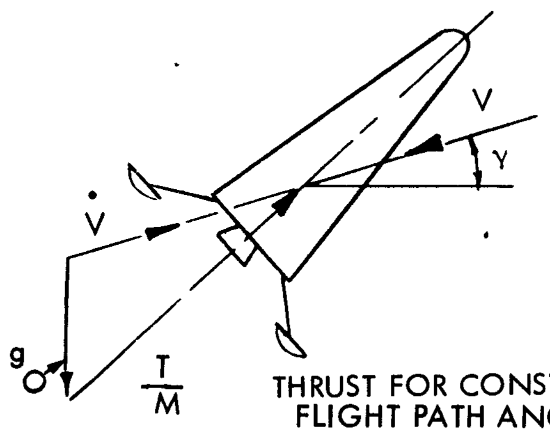
BALLUTE/RETROPROPULSIVE RETARDATION REQUIREMENTS

In this mode, the vehicle pulls up after entry and employs roll control to fly at a predetermined constant altitude. Full positive lift is applied prior to reaching equilibrium velocity to accomplish a zoom maneuver. The ballutes are deployed at the top of the trajectory, then jettisoned at a predetermined velocity-altitude combination; retropropulsion further decelerates the MEM at a constant flight path angle. Touchdown is accomplished after a period of hovering to select the final landing site.

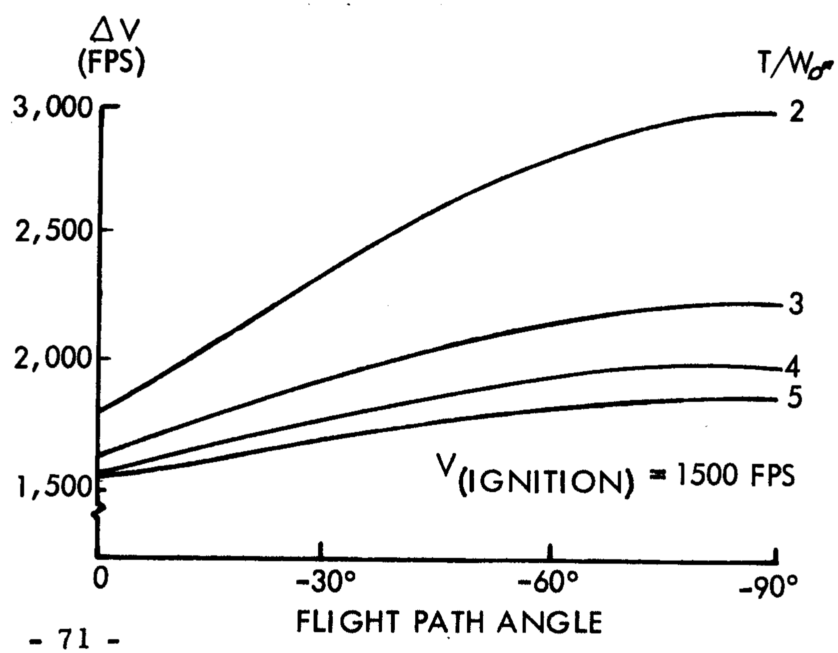
In order to achieve final deceleration by retropropulsion, the thrust must be applied in a direction which will simultaneously reduce the velocity vector and counteract the Martian gravity. The required thrust direction is indicated at the lower left. The retro ΔV required is shown as a function of the glide path angle and thrust-to-weight ratio. The steepest flight path angle at ballute jettison is approximately 30 degrees. At flight path angles of this magnitude and a thrust-to-weight (Mars) ratio of 4 (1.52 in Earth weight) the gravity losses are approximately 12 percent; this combination of spacecraft velocity at ignition, flight path angle, and T/W results in the minimum weight system when descent engine size and weights and propellant requirements are traded off.

BALLUTES/RETROPROPULSIVE RETARDATION REQUIREMENTS





THRUST FOR CONSTANT FLIGHT PATH ANGLE



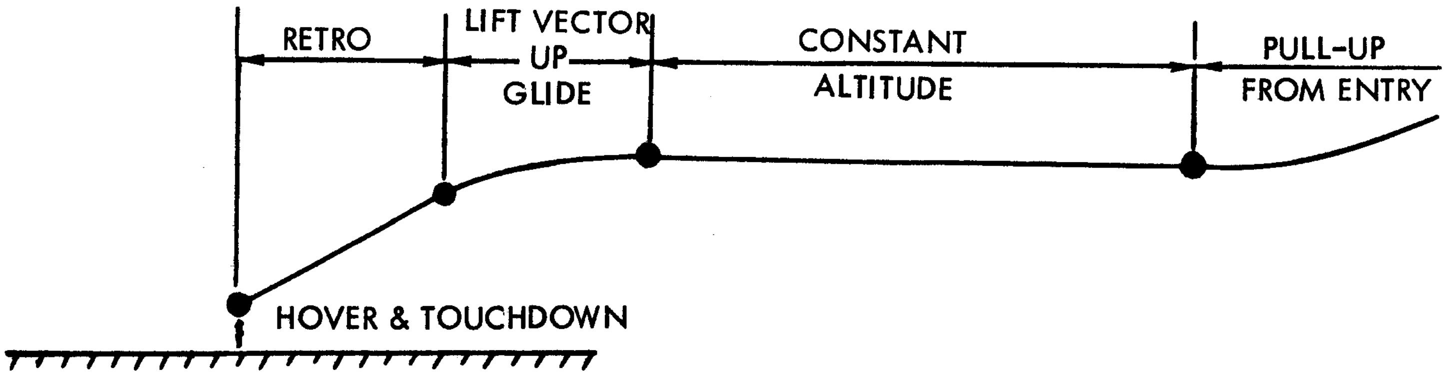
RETROPROPULSIVE RETARDATION REQUIREMENTS

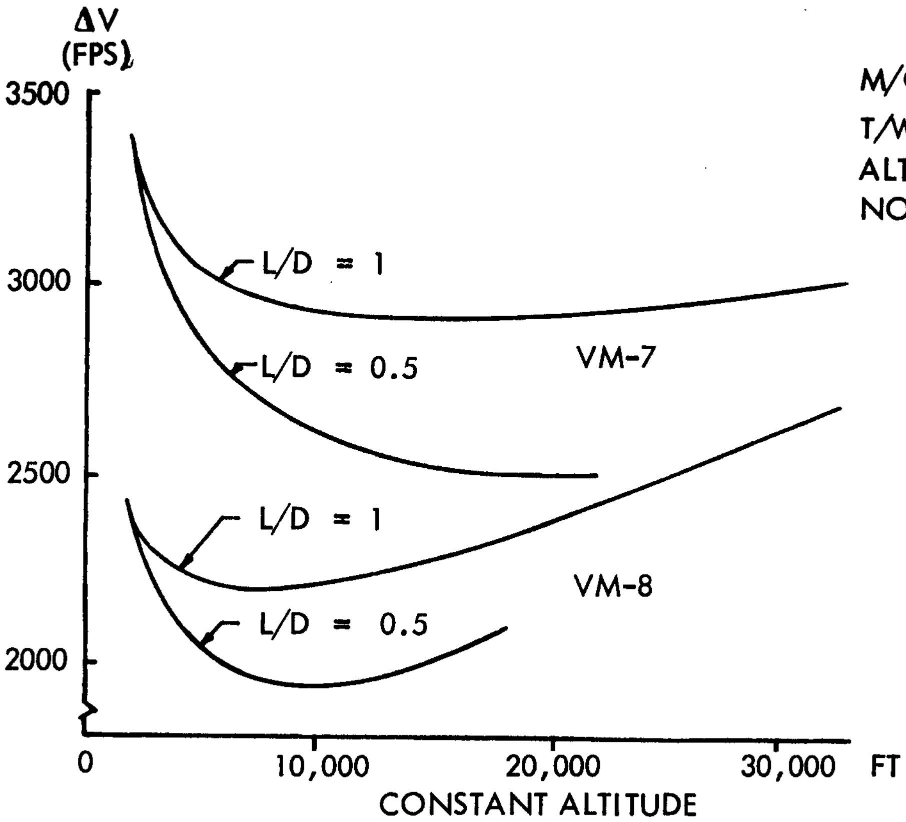
The corresponding flight mode for retropropulsive descent also uses a constant altitude leg following entry. In this case full positive lift is applied after equilibrium glide conditions are reached. The vehicle continues to glide in a gradually steepening flight path and the retropropulsion engine is ignited at an appropriate combination of velocity, altitude, and flight path angle. A constant flight path angle is maintained by directing the thrust as in the ballute/retropropulsion mode, and the maneuver terminates with a period of hover and touchdown.

The retropropulsive ΔV required is shown for a typical value of m/C_LA and a thrust to weight (Mars) ratio of 4.0 (1.52 in Earth weight) as a function of the altitude flown in the constant altitude leg. A wide range of pull-up altitudes can be accomodated for very little additional ΔV. Vehicles with L/D's of 0.5 require about 300 to 500 fps (90 to 150 m/sec) less ΔV than vehicles of L/D = 1 for the same m/C_LA values because the lower L/D vehicles exhibit more drag and therefore decelerate more before thrust application.

The optimum pull-up altitude is between 5,000 and 30,000 feet (1.5 and 9.1 km) in the VM-7 atmosphere, and 5,000 to 15,000 (1.5 to 4.6 km) in the VM-8 atmosphere. The VM-8 atmosphere is characterized by higher low-altitude densities and consequently lower equilibrium velocities. The ΔV required in VM-8 is approximately 600 fps (180 m/sec) less than for the VM-7 atmosphere.

RETROPROPULSIVE RETARDATION REQUIREMENT





$M/C_L A$ = 6.4 SLUGS/FT^2
T/W_{σ} = 4
ALTITUDE AT END OF BREAKING = 1000 FT
NO ALLOWANCE FOR HOVER & WIND

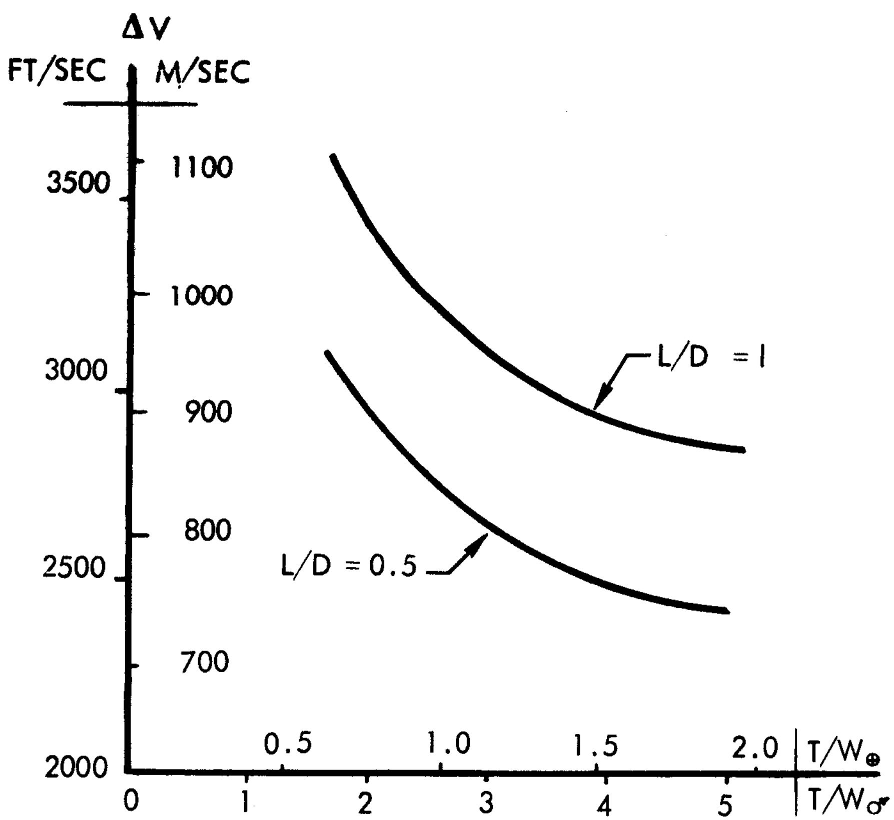
RETRO ΔV SENSITIVITY TO T/W AND M/C_LA

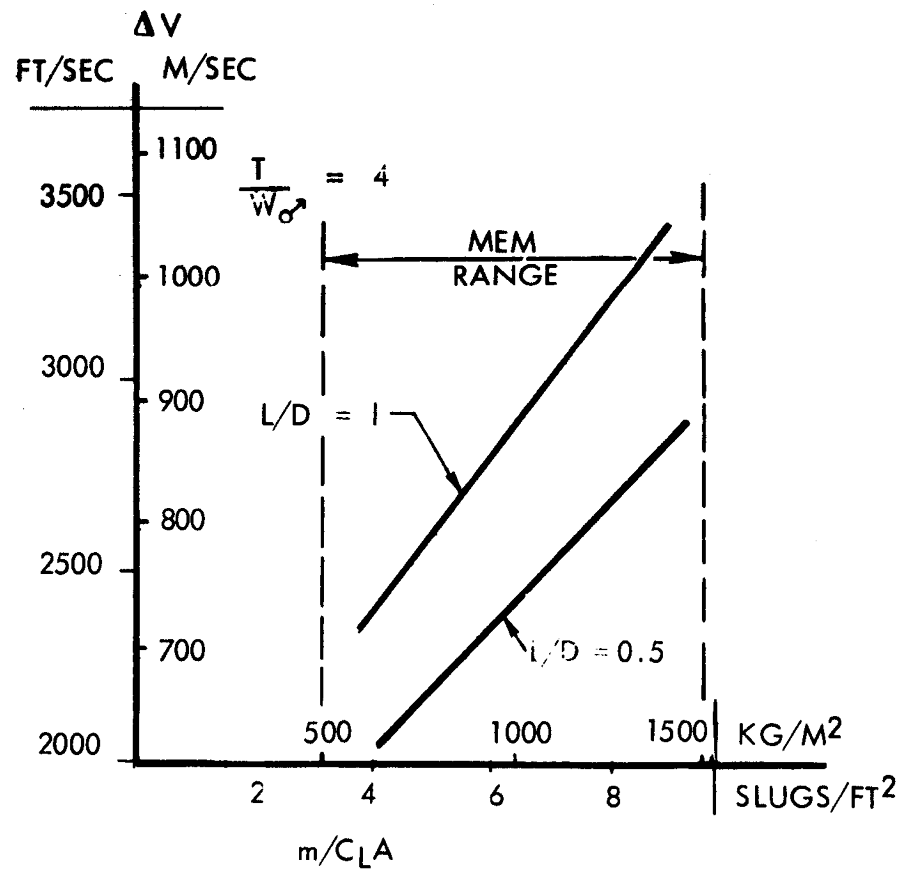
The ΔV required for the retropropulsive mode is quite sensitive to the thrust to weight ratio of the descent engine. Low thrust levels result in increased gravity losses because of the longer burn times; large thrusts result in large bulky engines. A practical compromise is a thrust to weight (Mars) ratio of 4 (1.52 Earth weight); only small reductions in ΔV can be achieved by larger ratios.

The required ΔV varies with m/C_LA because of the variation in initial equilibrium velocity at the optimum pull-up altitude. Approximately 150 fps (45 m/sec) more ΔV is required for an increase in m/C_LA of 1 $slug/ft^2$ in the range of interest (30 m/sec per 100 kg/m^2).

RETRO ΔV SENSITIVITY TO T/W & M/C_LA

VM-7 ATMOSPHERE





ΔV REQUIREMENTS FOR HOVER AND WINDS

The reserve propellant allowance for zero speed (after deceleration) is expressed in terms of hover capability. The hover time requirement was established at the beginning of the study as one to two minutes, or 750 to 1500 fps. This reserve propellant can be used to hover over the landing site to give the crew the opportunity for final inspection of the landing site, or to translate to an alternative landing site. The range that can be achieved is slightly under one nm for a one-minute hover time or 3.6 nm for two minutes.

The wind model provided by NASA indicated winds of up to 340 fps (104 m/sec) maximum in a storm; however, it is not considered likely or reasonable that a landing would be made in such conditions. A ΔV allowance of 250 fps (76 m/sec) was included to counteract winds in the same direction as the vehicle's flight. No allowance was made for gusts since they generally will have very small effects on the vehicle motion because of the low density.

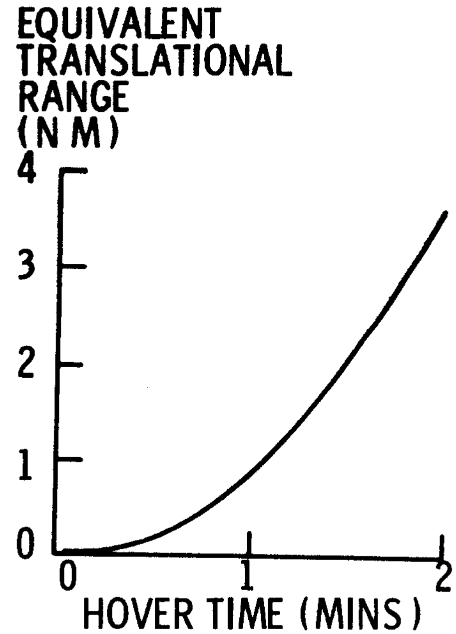
Typically, with a wind speed of 170 fps (52 m/sec) in VM-7 atmosphere, the spacecraft can hover over a landing site with a pitch angle of approximately one degree to the vertical; the horizontal component of the thrust then balances the aerodynamic drag.

ΔV REQUIREMENTS FOR HOVER AND WINDS

HOVER

1 TO 2 MINS

1 MIN - ΔV = 750 FPS
2 MIN - ΔV = 1500 FPS



WINDS

AVERAGE	0 - 85 FPS
AVER. STORM	170FPS
MAX. STORM	340FPS

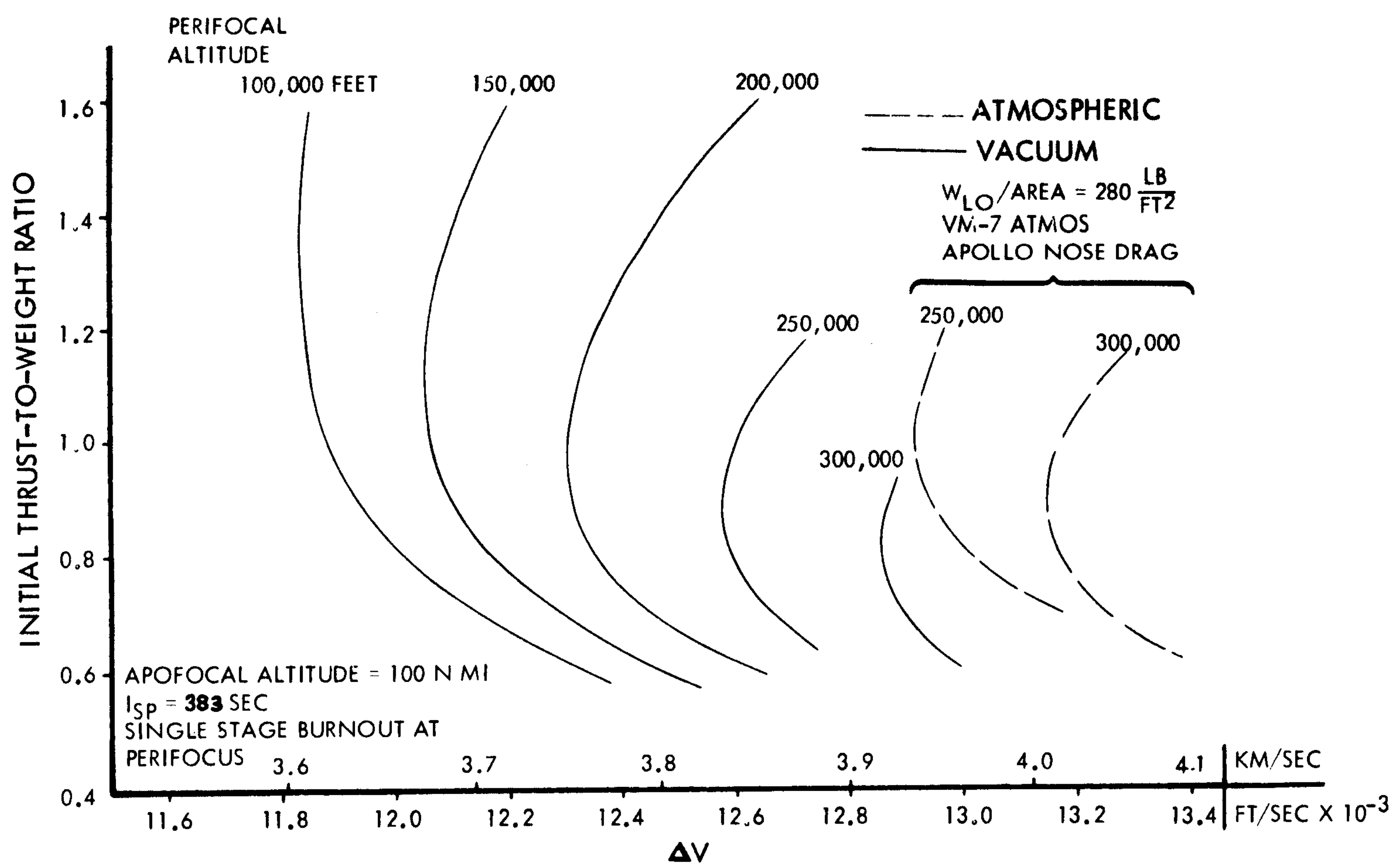
ALLOW ΔV = 250 FPS
NO LANDING IN MAX. STORM

ASCENT $T/W_{\oplus}$ REQUIREMENTS

LIFTOFF TO BURNOUT

The MEM ascends from the Mars surface to an intermediate 100-NM (185-km) circular orbit to afford proper phasing for rendezvous with the orbiting spacecraft. The relationship between liftoff thrust-to-weight ratio and required ΔV is shown for a number of burnout perifocal altitudes for vacuum and atmospheric trajectories assuming a liftoff weight-to-area ratio of 280 lb/ft^2 (1370 kg/m^2). The largest drag losses are experienced in the VM-7 atmosphere because of its relatively high density at high altitudes. Minimum ΔV requirements occur with thrust-to-(Earth) weight ratios of 0.8 to 1.0. Consideration of the additional drag losses during coast to apoapsis at 100 NM (185 km) indicates that a periapsis altitude of 300,000 ft (91 km) requires the minimum ΔV for the VM-7 atmosphere.

ASCENT T/W$_\oplus$ REQUIREMENTS - LIFT-OFF TO BURNOUT



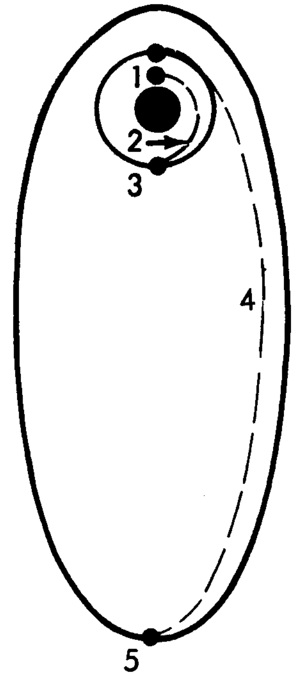
ASCENT ΔV REQUIREMENTS

Typical ascent ΔV budgets are shown for the low circular and two elliptical orbit (e = 0.6 and 0.9) missions. The ascent mode consists of (1) an initial burn to 300,000 ft (19 km); (2) coast up to 100 NM (185 km); (3) circularize at this altitude with a short burn; (4) after an appropriate time to obtain proper phasing with the spacecraft, ascend for rendezvous; and (5) rendezvous with the spacecraft at apoapsis.

The drag losses and the allowance for non-equatorial inclinations and circularization are relatively small for all three missions. However, the energy required for transfer to the final orbit increases with orbit eccentricity; ΔV requirements vary from approximately 16,000 fps (4.9 km/sec) for the low circular orbit to 20,350 fps (6.2 km/sec) for the elliptical orbit of 0.9 eccentricity. A contingency allowance of 10-percent is included in these figures to allow for a launch window and abort capability.

ASCENT ΔV REQUIREMENTS

EVENT	ORBIT		
	270 NM CIRCULAR	162 x 6140 NM (e = 0.6)	162 x 36,100 NM (e = 0.9)
1. ASCENT TO h=300KFT			
IN VACUUM, i = 0	12,800 (FPS)	12,800 (FPS)	12,800 (FPS)
INCLINATION EFFECT	520	780	780
VM-7 DRAG LOSSES	300	300	300
2. COAST TO 100 NM VM-7 DRAG LOSSES	180	180	180
3. CIRCULARIZE 100 NM	75	75	75
4. & 5. TRANSFER TO FINAL ORBIT	550	3,100	4,352
	14,425	17,235	18,487
10% CONTINGENCY	1,443	1,724	1,849
TOTAL ΔV	15,868	18,959	20,336

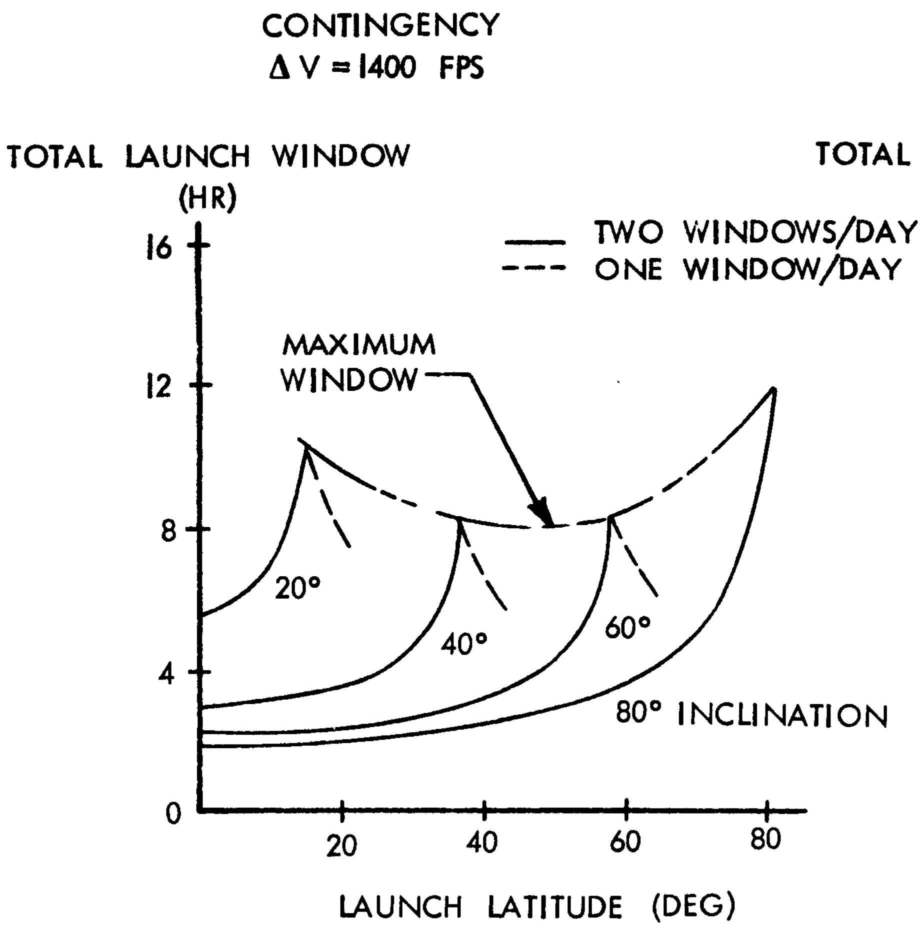


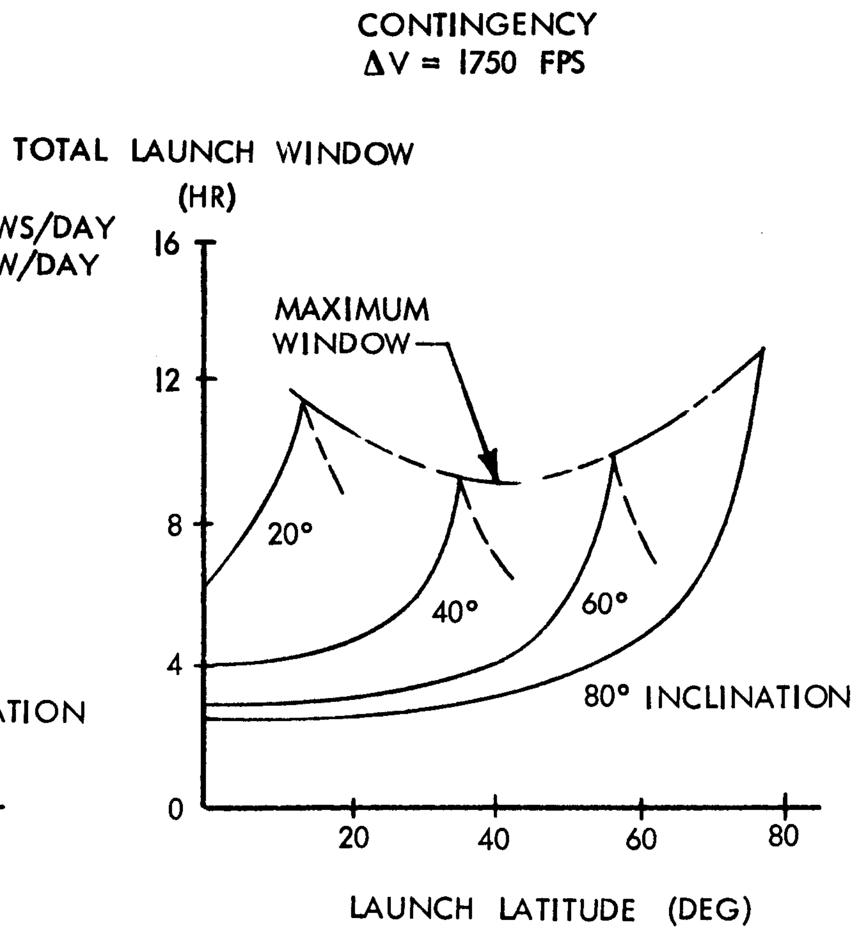
DAILY LAUNCH WINDOW

If the latitude of the launch site on a rotating planet is less than the inclination of the desired orbital plane, two launch opportunities per day are available when the launch site crosses the orbital plane. The 10-percent contingency allowance in ascent ΔV over that required for a pure coplanar ascent permits the MEM to be launched slightly before or after the nominal launch time. The excess ΔV may be employed to correct the launch ascent plane at the time the vehicle intersects the desired orbit plane. Typical MEM launch windows are shown as a function of latitude and orbit inclination for ascent ΔV contingency allotments of 1400 fps (426 m/sec) and 1750 fps (533 m/sec). If the angular difference between the launch latitude and the orbital inclination is approximately 5 degrees, the two daily windows overlap and result in a continuous window as shown by the dashed curves. Maximum window size occurs for either low- or high-inclination/latitude combinations but in all cases significant windows in the 8- to 10-hour class exist if the MEM orbit plane and launch site latitude are correctly matched.

DAILY LAUNCH WINDOW

LAUNCH TO 100 NM CIRCULAR ORBIT





MAXIMUM LAUNCH WINDOW WHEN LATITUDE = INCLINATION -5°

TYPICAL MISSION ABORT PENALTIES

The inclination of the spacecraft orbit is chosen as a function of the planned arrival and departure dates chosen so that the departure orbit is coplaner. Early or late departures generally entail a spacecraft ΔV penalty which is comprised of two major elements: (1) the plane change penalty to enter the required departure plane and (2) the difference in departure injection ΔV due to the varying departure hyperbolic excess velocity (V_{∞}), which is a function of departure date. For elliptical parking orbit, the location of periapsis for both the arrival and departure conditions also must be considered.

In the example shown for a 1984 retrobraker direct mission with a nominal stay time at Mars of 40 days, larger variations in the spacecraft ΔV requirements occur for some orbits. In some cases, the effect of V_{∞} variation is stronger than the plane change penalty and an early abort may result in a savings in required ΔV since the cost of the 40-day stay itself is relatively high in the sense of higher V_{∞} values. Under certain conditions, early departure may be quite expensive, and provision for a complete abort ΔV contingency budget may be prohibitive.

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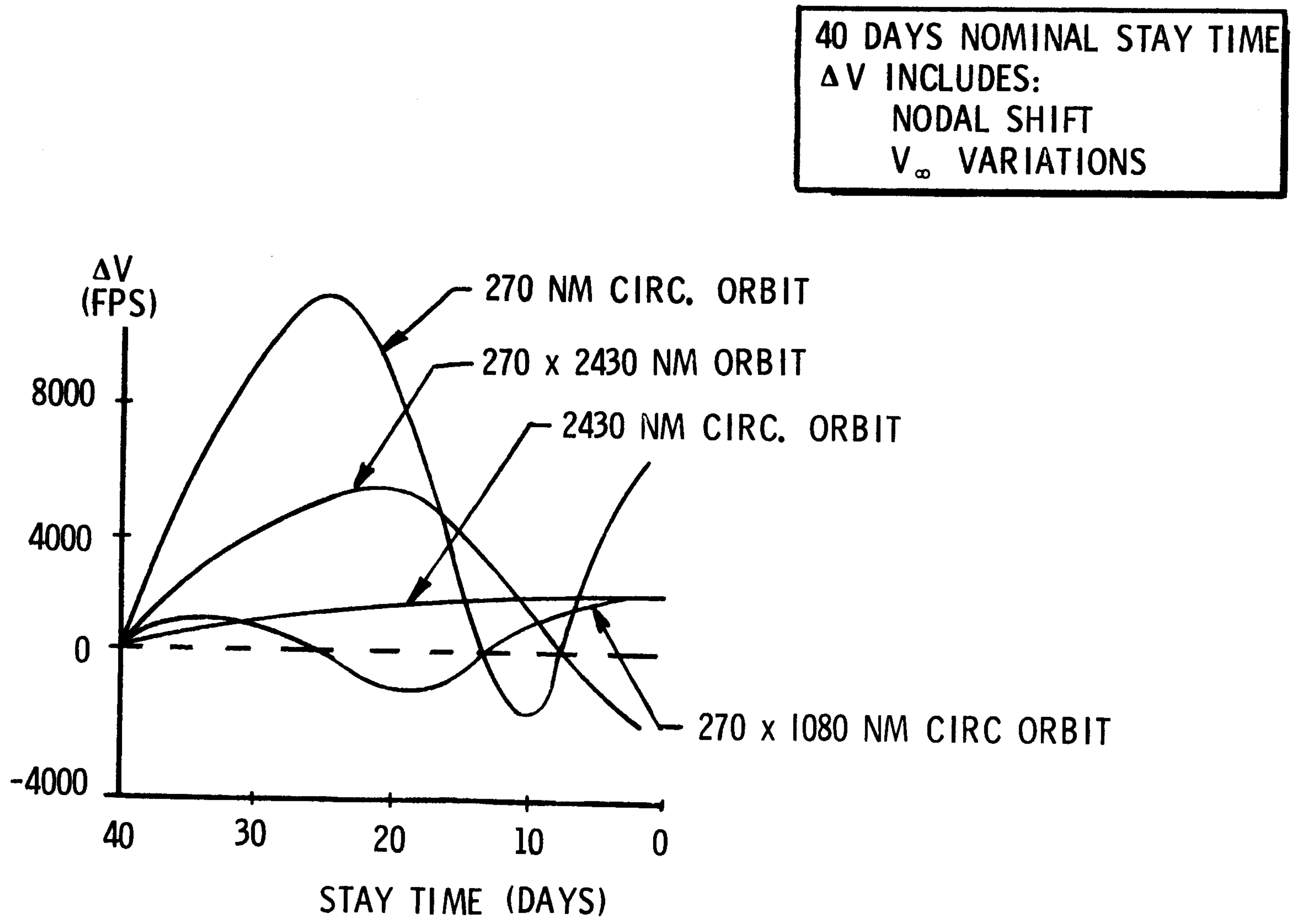
TYPICAL MISSION ABORT PENALTIES
(1984 R/B DIRECT MISSIONS)

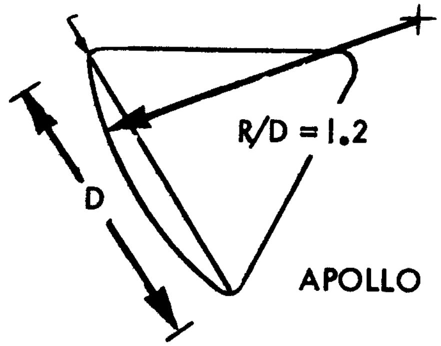


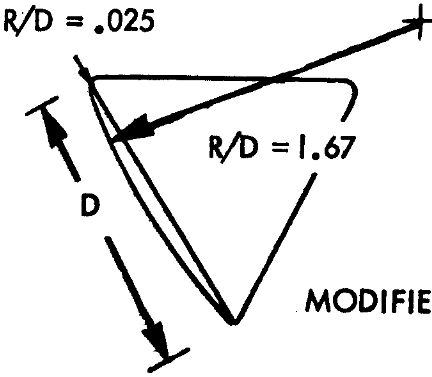
AERODYNAMIC CHARACTERISTICS

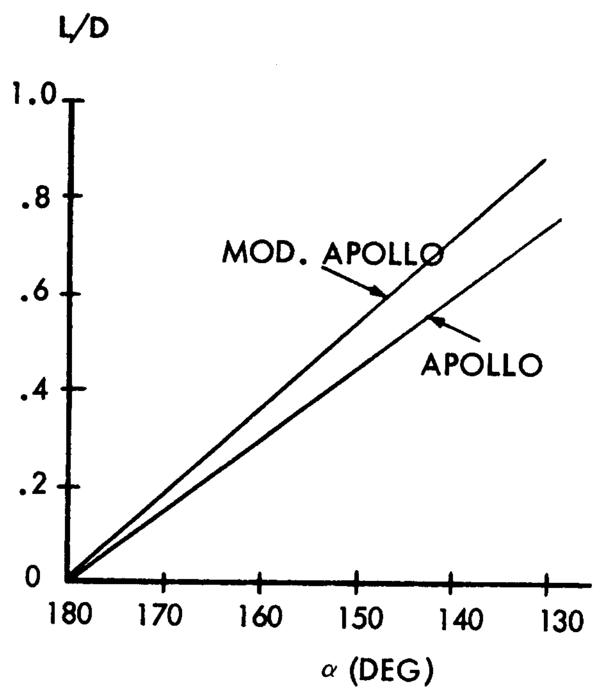
LOW L/D CONFIGURATION

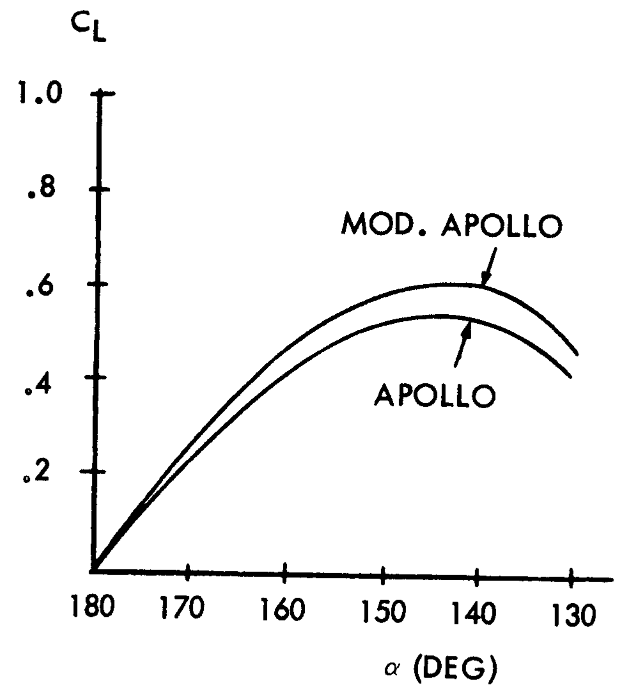
An Apollo shape was selected for the low L/D MEM configuration. The less severe Martian entry environment (as compared to Earth entry heating) allow aerodynamic optimization of the vehicle shape to obtain maximum lift. The magnitudes of C_L and L/D at a given angle of attack may be increased by increasing the main heat shield radius from 1.2 to 1.67 times the diameter (resulting in a blunter shape). The lift coefficient may be further increased by reducing the corner radius of the body from 0.05 to 0.025 times the diameter without any adverse effect on static stability characteristics. This results in a lift coefficient of 0.604 at a 147 degree angle of attack, with a lift-to-drag ratio of 0.568. Although these changes degrade the transonic and subsonic stability characteristics, this effect is not pertinent because the MEM retardation and/or retropropulsion is sequence (i.e., ballutes or retropropulsive descent) is initiated above Mach 3.

AERODYNAMIC CHARACTERISTICS
LOW L/D CONFIGURATION









TRANSONIC STABILITY NOT REQUIRED

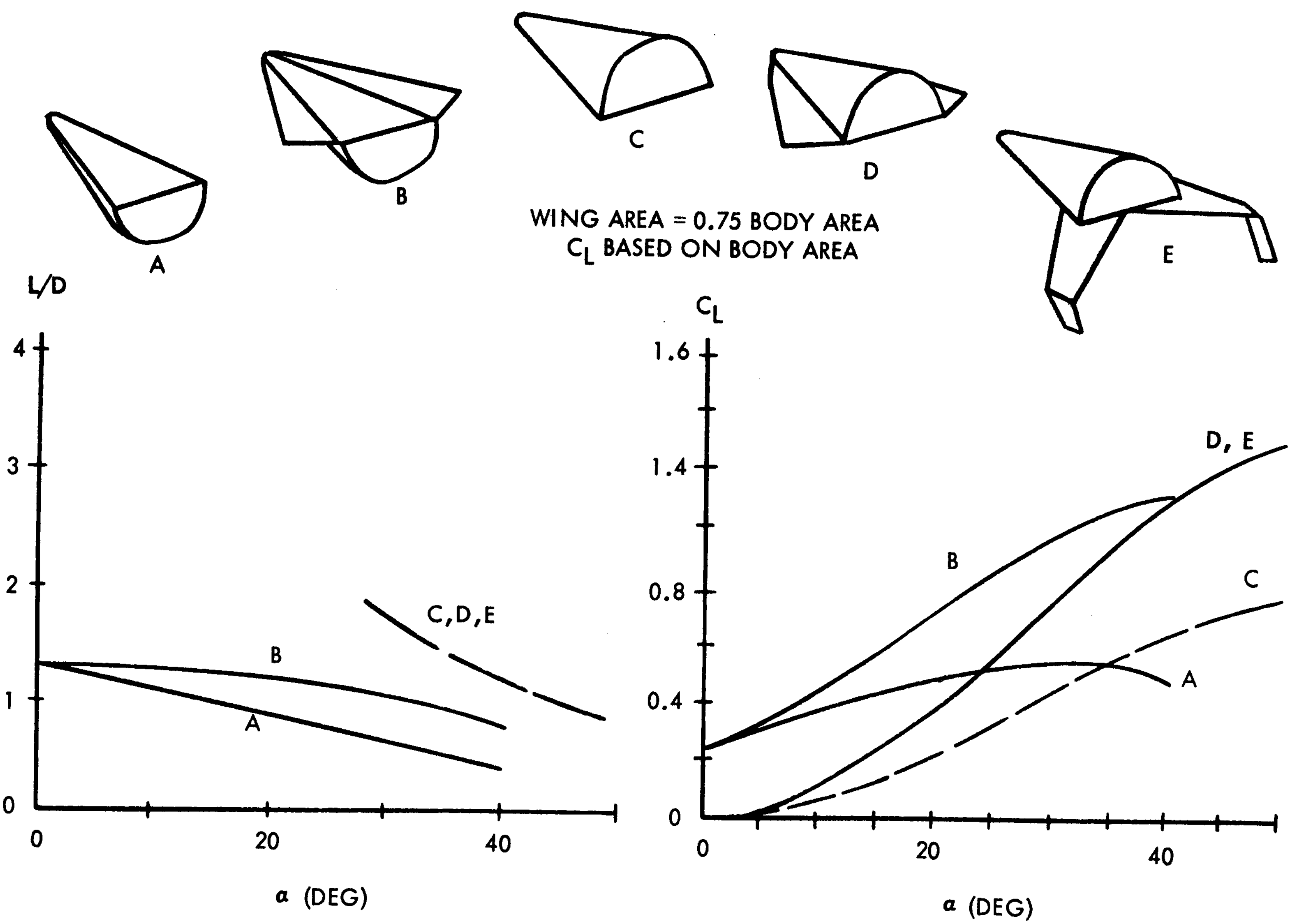
AERODYNAMIC CHARACTERISTICS

LIFTING BODY CONFIGURATION

A number of aerodynamic configurations were considered for the lifting body. The basic body shape consists of a blunted half cone with an 18-degree half-cone angle. Variations included flying both flat side up and flat side down and the use of folding wings to increase the lift. The wings shown in sketches B and D fold along the edges of the half cone; those at E have transverse hinge lines across the rear end of the vehicle and result in more acceptable center-of-gravity requirements. Lift-to-drag ratios of 1.0 can be obtained at angles of attack of 20 degrees for the basic flat side up configuration or at $\alpha = 40$ degrees for the flat-side-down version.

The selected configuration achieves a lift coefficient of 0.63 at $\alpha = 40$ degrees and an L/D value of 1.0. This results in a higher C_L than the flat-side-up version at the desired L/D. Wings ultimately were rejected since the advantages accruing from the increase in lift are largely cancelled by the additional weight and complexity of the system. Longitudinal stability is obtained by a 5-degree boat tail at the rear 20 percent of the vehicle.

AERODYNAMIC CHARACTERISTICS
LIFTING BODY CONFIGURATION

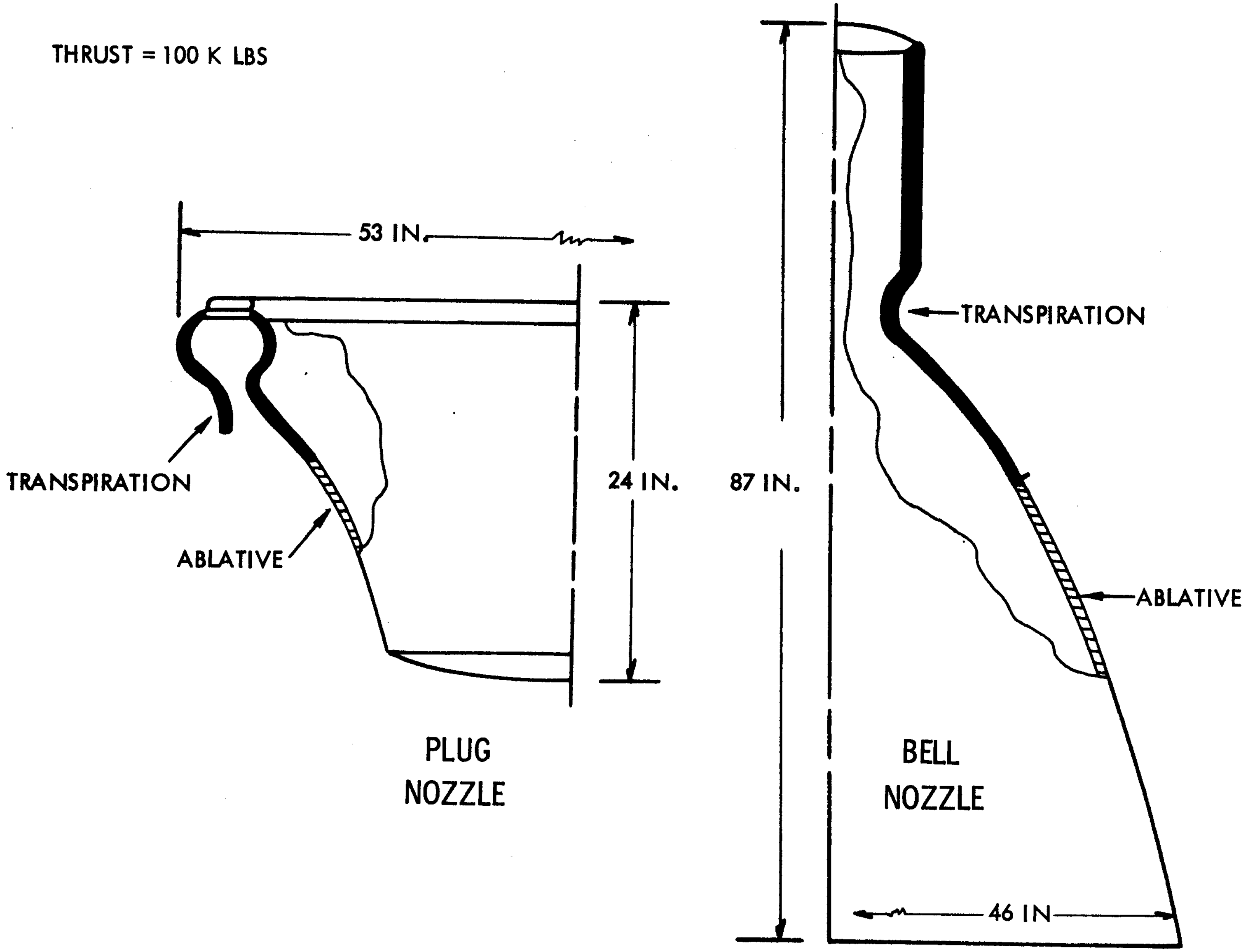


ENGINE CONFIGURATIONS

Schematic sketches are presented for 100,000-pound (45,000 kg) thrust, transpiration-cooled, plug and bell nozzle engines. Although the overall dimensions of plug nozzles are smaller, the relative areas which must be transpiration-cooled (i.e., for space storable propellants) are greater than for the bell nozzles; heating rates at plug nozzle throats also are greater because of the shorter gas flow paths. These factors contribute to the higher cooling requirements of plug as compared to bell nozzle engines.

ENGINE CONFIGURATIONS

THRUST = 100 K LBS



PLUG NOZZLE

BELL NOZZLE

DELIVERED SPECIFIC IMPULSE

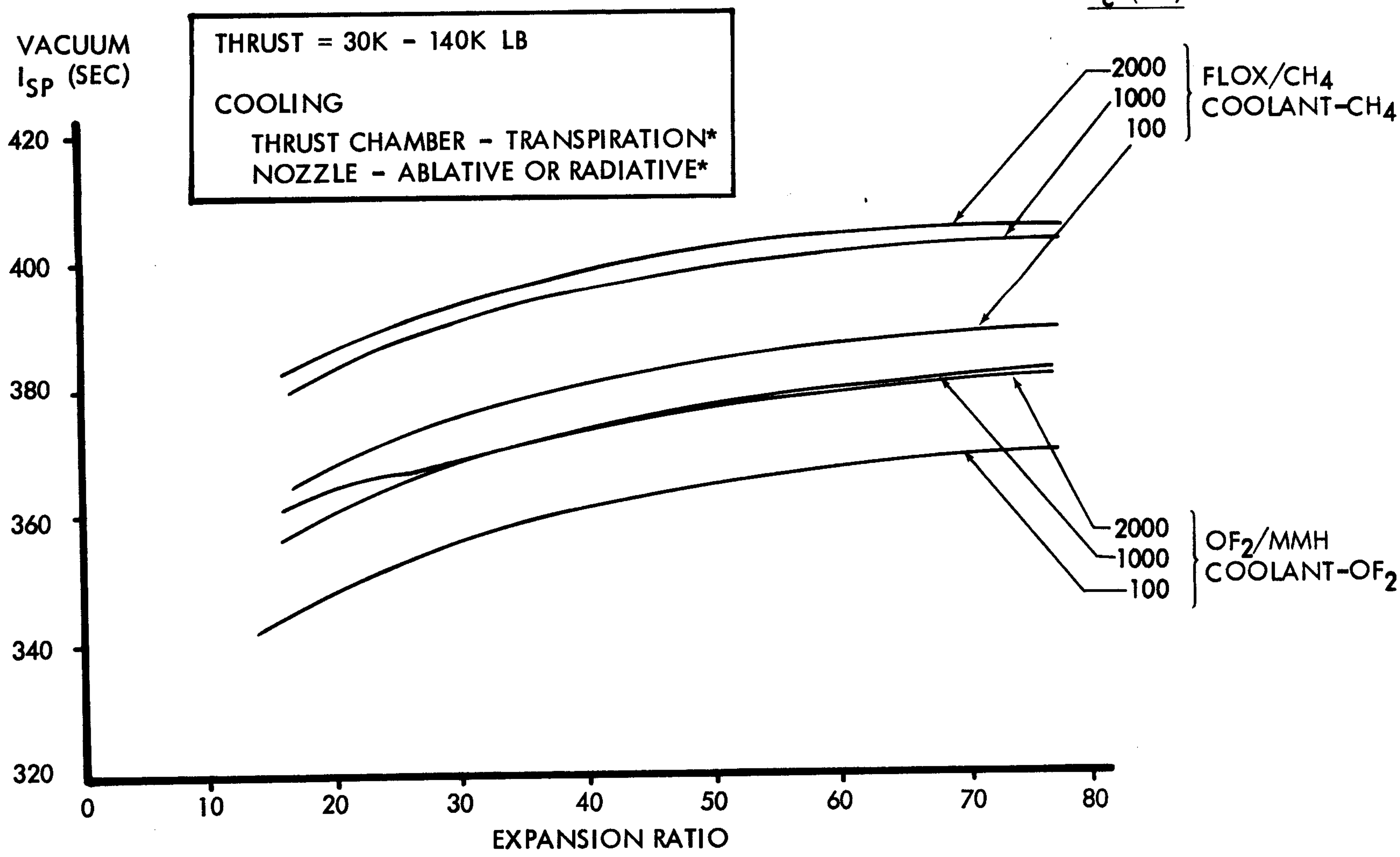
BELL NOZZLES

The performance of composite bell nozzle engines which employ a combination of transpirative, ablative, and radiative cooling techniques was determined. Both $FLOX/CH_4$ and OF_2/MMH propellant combinations were considered with CH_4 and MMH, respectively, acting as the coolant.

The specific impulse of OF_2/MMH is approximately 20 to 25 seconds lower than for $FLOX/CH_4$. Since ablative cooling can be used beyond an expansion ratio of 14 at chamber pressures of 1,000 psi (70 kg/cm^2) or less, transpiration cooling losses are lower at this pressure level. At pressures of 2,000 psi (140 kg/cm^2) transpiration cooling must be used up to an expansion ratio of 30. Further losses therefore are encountered when OF_2 is used as the transpirant, thus accounting for the reduction in delivered performance at higher chamber pressures.

All losses, including those due to transpiration up to the critical nozzle area, were considered. It was found that the specific impulse is essentially independent of engine size in the range of 30,000- to 140,000-pound (13,600- to 63,600- kg) thrust.

DELIVERED SPECIFIC IMPULSE
BELL NOZZLES



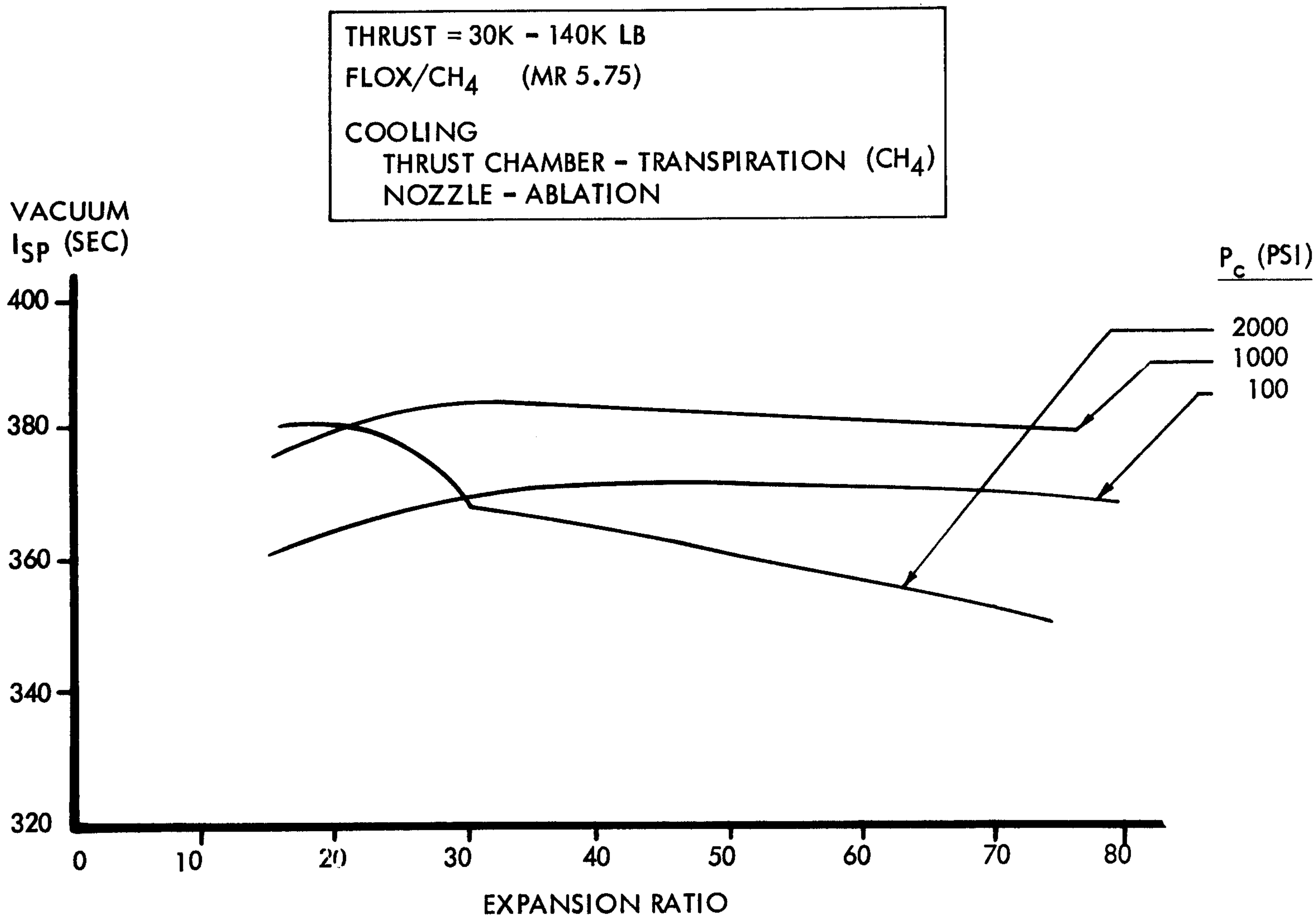
*TRANSITION VARIES WITH P_c

DELIVERED SPECIFIC IMPULSE PLUG NOZZLES

The performance of plug nozzle engines employing both transpiration and ablation cooling also were estimated. The performance of a plug nozzle with FLOX/CH_4 as the propellant combination and CH_4 as the transpirant is significantly lower than that obtained with a bell nozzle because of the higher transpirant flow rates required, particularly at high chamber pressures (e.g., 2,000 psi, 140 kg/cm^2). (These higher flow rates are required because of the relatively large surface areas which must be cooled in the plug nozzle.) FLOX/CH_4 will deliver an impulse of 383 seconds at a chamber pressure of 1,000 psi (70 kg/cm^2) and expansion ratios ranging from 30 to 40. By comparison, the poor cooling capability of OF_2 as a transpirant reduces the specific impulse of OF_2/MMH in plug nozzle engines to the order of 330 seconds.

The confidence in these aerospike performance estimates is considerably less than that for the bell nozzle engines. Analytical and experimental research is needed to define the aerospike engine design limitations and delivered performance with space storable propellants.

DELIVERED SPECIFIC IMPULSE
PLUG NOZZLES



RETARDATION AND LANDING SYSTEM WEIGHT

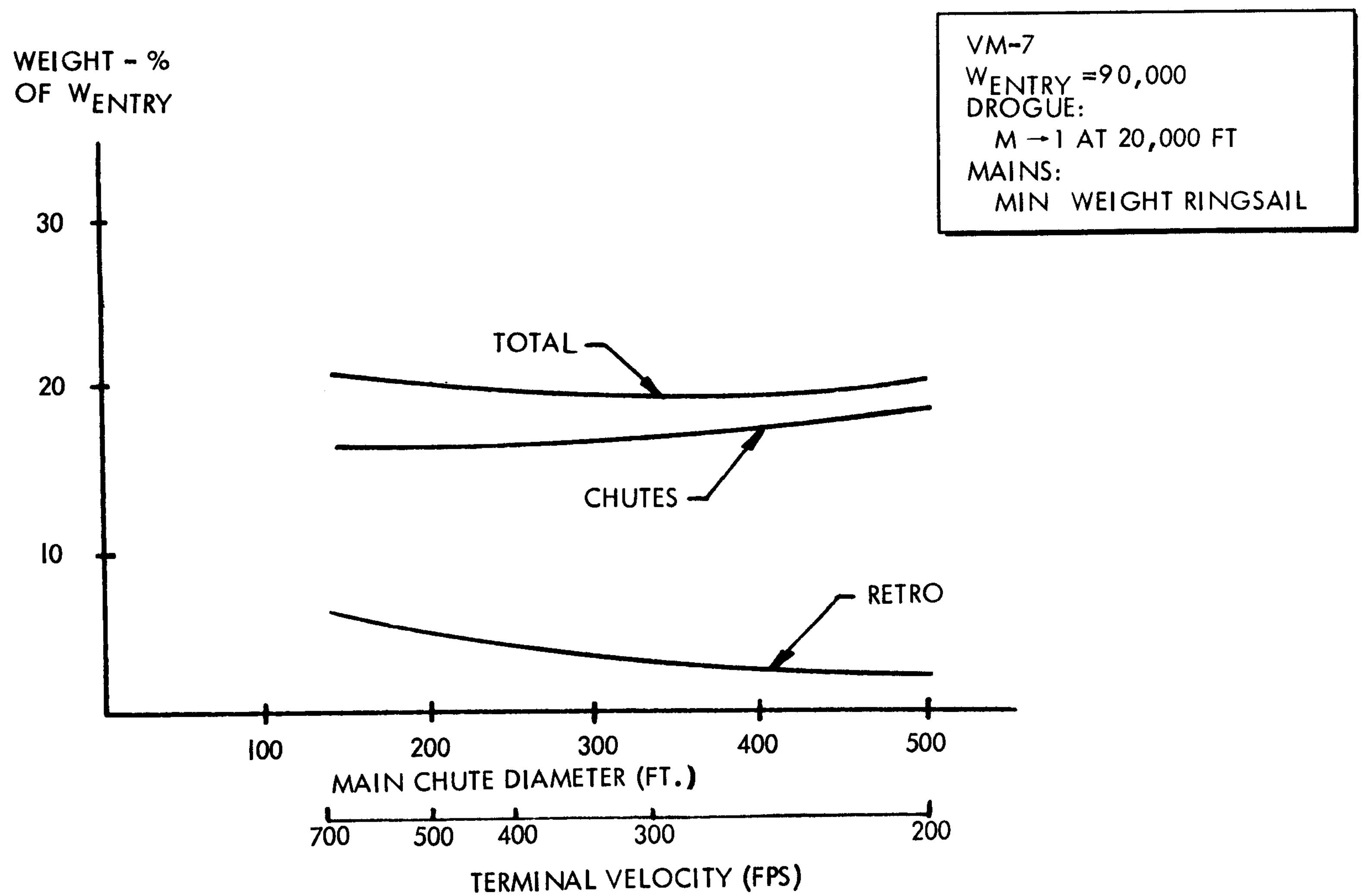
TWO-STAGE CHUTE/RETRO SYSTEM

Optimization of a parachute/retropropulsion system includes trading off parachute size (and terminal velocity) against the retro-propulsion requirements. Typical variations of the weight fraction of parachute and retropropulsion system weights are shown for a two-stage supersonic hyperflo-type drogue and large ringsail or cloverleaf main parachutes. Redundancy is assumed for both stages.

The parachute weight fraction increases slowly with the diameter of the main chutes while the propulsion weight decreases (because of the lower terminal velocity). Minimum total system weight occurs with a main chute diameter of 400 ft (120 m) at 18 percent of the entry weight. The terminal velocity of the main chutes is approximately 250 fps (76 m/sec). Smaller parachute sizes, e.g., 160 ft (50 m), may be more practical, and would result in terminal velocities of 600 fps (183 m/sec) and a total system weight fraction of 21 percent.

RETARDATION AND LANDING SYSTEM WEIGHT

2-STAGE CHUTE/RETRO SYSTEM



RETARDATION AND LANDING SYSTEM WEIGHT

SINGLE-STAGE BALLUTE/RETRO SYSTEM

An appreciably lighter and simpler retardation and landing system results when the second-stage main parachute is eliminated. A single-stage redundant ballute, which results in a minimum weight fraction of 14 to 16 percent (allowing for weight uncertainties), was selected. A ballute diameter of 60 ft (18.3 m) would be indicated, and the resulting terminal velocity would be 1400 fps (430 m/sec).

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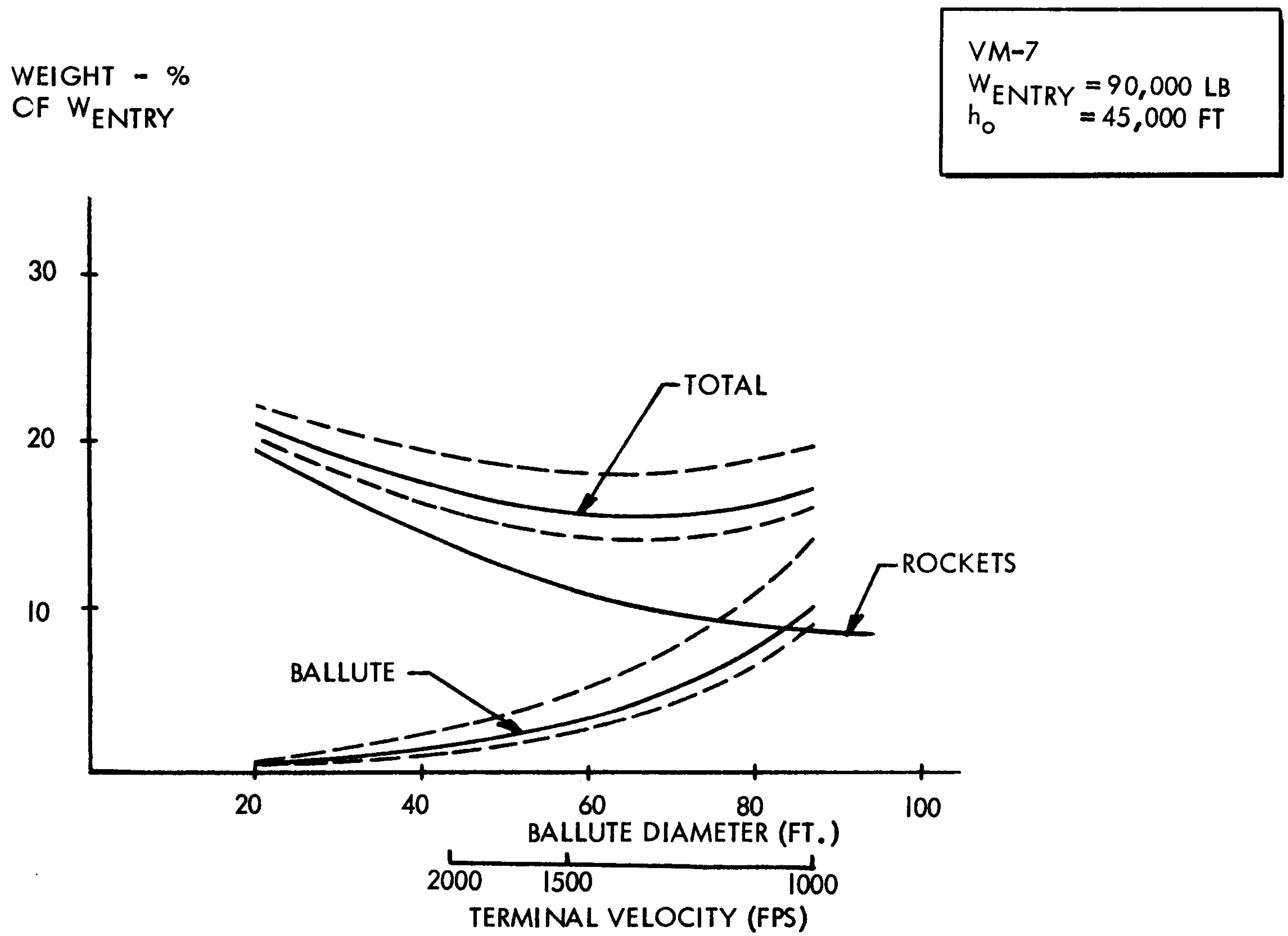
RETARDATION AND LANDING SYSTEM WEIGHT

SINGLE STAGE BALLUTE/RETRO SYSTEM



TYPICAL BALLUTE SUBSYSTEM CHARACTERISTICS

A typical ballute arrangement for a 30-ft (9.1-m) diameter low L/D MEM is shown. The ballute is nominally 60 ft (18.3-m) in diameter (exclusive of the burble fence located at its maximum dimension to enhance aerodynamic stability). The ballute is inflated by ram air.

Each of the two ballutes in the system weighs 980 lb (444 kg). The total subsystem weight, including the structural scar weight to accept the concentrated loads, bags, mortars, pilots, and sequencers is estimated at 3270 lb (1483 kg). Deployment takes place at Mach 3.5 (approximately 3500 fps, 1070 m/sec) at an altitude of 30,000 ft (9.1 km) in VM-7 while flying horizontally at the top of a "zoom" maneuver. Although the velocity is high, the dynamic pressure is only 50 psf (243 kg/m^2), and the heating rates are low enough to allow the use of neoprene-coated nomex for the envelope. Terminal conditions reached at the time the ballute is jettisoned and the retropropulsion engine is ignited are Mach 1.4 at 10,000 ft (~3.0 km) altitude with a flight path angle of -22 degrees.

TYPICAL BALLUTE SUBSYSTEM CHARACTERISTICS

WEIGHTS

BALLUTE WEIGHT

ENVELOPE, NOMEX	196 LB
COATING, NEOPRENE	148
MERIDIAN CABLES	344
RISER	292
	980

SUBSYSTEM WEIGHT

2 BALLUTES	1960
STRUCTURE, BAGS, MORTARS, ETC.	1310
	3270

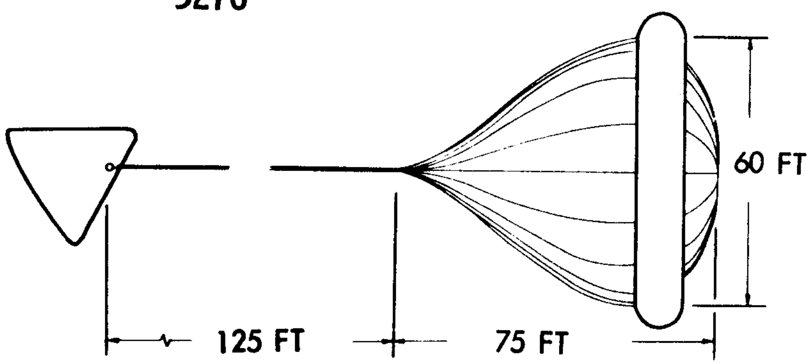
PERFORMANCE (VM-7)

DEPLOYMENT CONDITIONS

$M = 3.5$
$h = 30{,}000$ FT
$\gamma = 0$
$q = 50$ PSF

TERMINAL CONDITIONS

$M = 1.4$
$h = 10{,}000$ FT
$\gamma = -22^\circ$



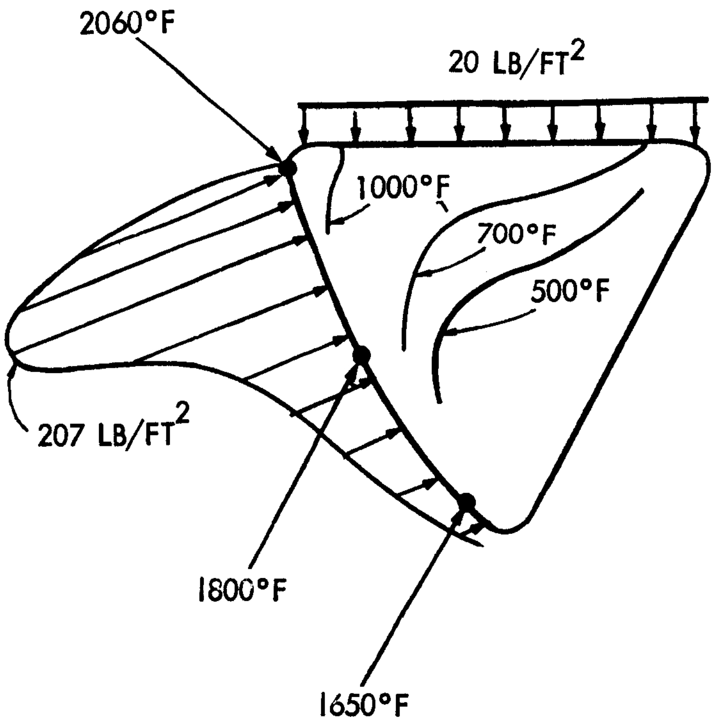
STRUCTURAL LOADS AND TEMPERATURE DISTRIBUTIONS

Typical pressure and temperature distributions for a low circular orbit entry are presented. Pressure levels generally are low; peak accelerations are on the order of one g. Equilibrium radiation temperatures for these entry conditions are 2060 F (1400 K) at the stagnation point of the low L/D vehicle and drop off rapidly to 1800 F (1260 K) at the center of the blunt face and to below 1000 F (810 K) over most of the conical body. The latter temperatures determine the limits at which ablators L605, and titanium are used.

STRUCTURAL LOADS AND TEMPERATURE DISTRIBUTIONS

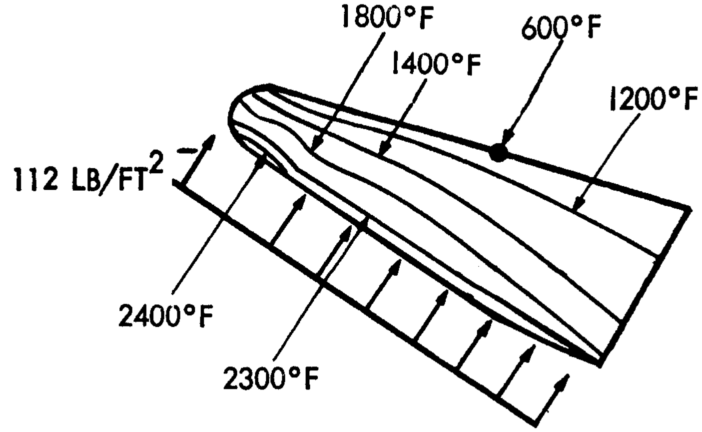
VM-7
$V_E = 11{,}050$ FPS
$\gamma_E = -7°$
$M/C_L A = 6.4$ SLUGS/FT^2

LOW L/D



PEAK ENTRY LOAD
= 1.2 g

LIFTING BODY



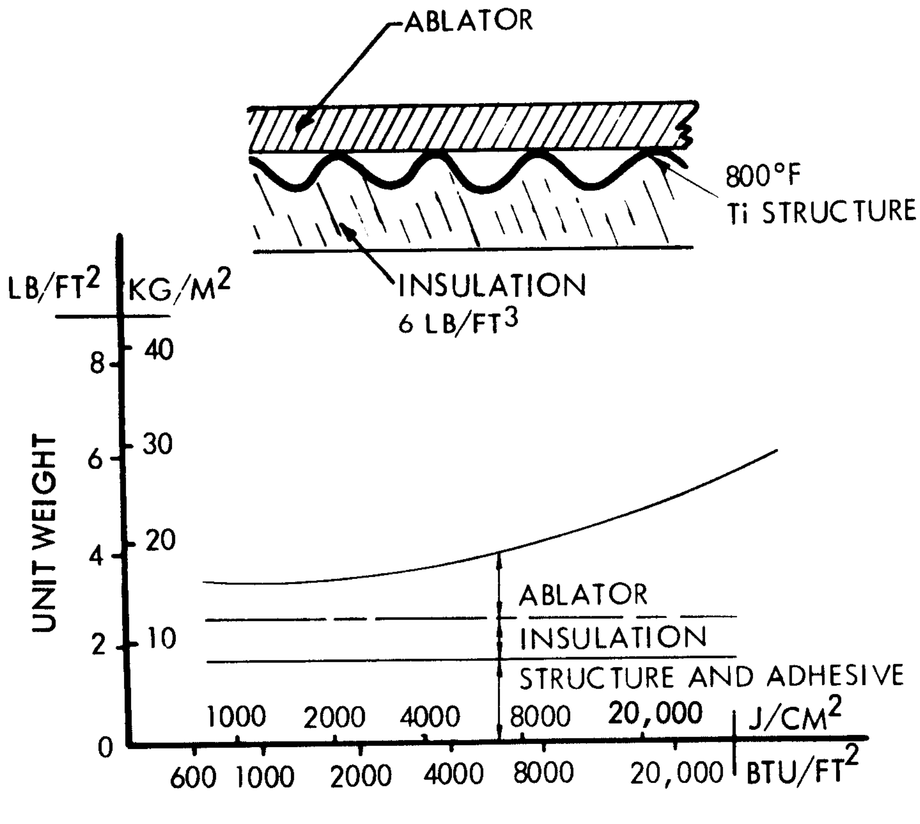
PEAK ENTRY LOAD
= 0.9 g

TYPICAL THERMOSTRUCTURAL WEIGHTS

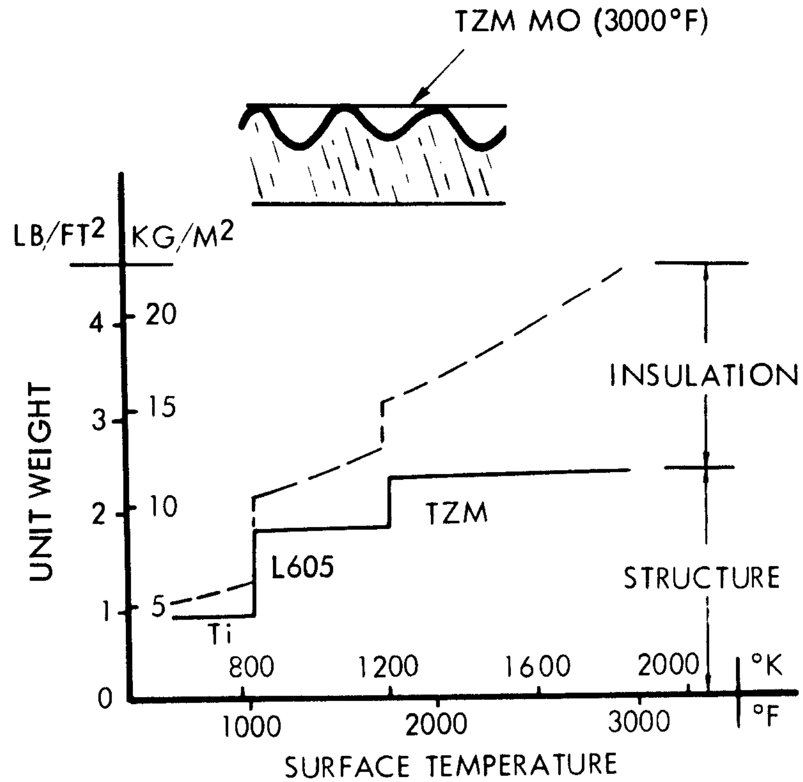
Results of a parametric evaluation of thermostructural unit weights are shown. In the data on the left, Avcoat 5026-39 ablator on a titanium corrugated substructure with TG-15000 insulation material was considered. The ablator was sized to maintain an 800 F (700 K) bondline temperature (projected for the MEM time period), and the insulation was sized for 150 F (340 K) at touchdown. Thermostructural weights of 3 to 6 lb/ft^2 (15 to 30 kg/m^2) result over a wide range of heat loads.

The figure on the right shows unit weights for a radiative-type structure as a function of peak temperature. The three materials considered were titanium up to 1000 F (810 K), L605 up to 1800 F (1260 K), and TZM molybdenum up to 3000 F (1920 K). Dyna-quartz insulation will maintain its integrity to 3000 F (1920 K). Minimum gauge materials generally are needed so that the weight variation is entirely due to the insulation requirements within each temperature range. Unit weights vary from 1.0 lb/ft^2 (5 kg/m^2) at 800 F (700 K) using titanium, to 2.8 lb/ft^2 (14 kg/m^2) at 1800 F (1260 K) using L605, and to nearly 5 lb/ft^2 (25 kg/m^2) at 3000 F (1920 K) using TZM.

TYPICAL THERMOSTRUCTURAL WEIGHTS



ABLATIVE THERMOSTRUCTURE



RERADIATIVE THERMOSTRUCTURE

ENVIRONMENTAL CONTROL & LIFE SUPPORT SUBSYSTEM

Analysis of the environmental control and life support subsystem (ECLSS) defined the most suitable systems from the viewpoint of performance, weight, and technology status for the 2-man/4-day and 4-man/30-day vehicles. Various concepts for gas storage, CO_2 absorption, and water recovery were analyzed in selecting the ECLSS subsystem.

Cryogenic storage of the breathing gases (2-gas system) was selected because of the lighter tankage and smaller volume requirements. Insulation techniques similar to those for cryogenic propellants and fuel cell reactants are applicable. Oxygen recovery was not considered feasible because of the additional system complexity, weight, and short-mission duration. Lithium hydroxide was selected for CO_2 absorption for the 2-man 4-day MEM. A two-bed molecular sieve was chosen for the 4-man/30-day MEM because of its weight, power, and volume advantage over a four-bed molecular sieve.

Storage is recommended for the water management system on the 2-man/4-day mission because of its simplicity. Use of a multifiltration process for the recovery of wash water and condensates would result in a weight savings of only 65 pounds (30 kg).

Since 420 pounds (190 kg) of stored water would be required for the 4-man/30-day mission, even though a 2 KW_e fuel cell is used, a multifiltration process was selected to supplement the fuel cell water output. If fuel cells are not used, a vacuum distillation process will be required to provide the necessary water through recovery of urine, wash water, and condensates. In addition, a four-bed molecular sieve would be used to permit recovery of the water absorbed by the silica gel. The total weight of the ECLSS is 386 to 481 pounds (175 to 218 kg) for the 2-man/4-day mission or 1544 to 1724 pounds (700 to 783 kg) for the 4-man/30-day mission, depending on whether fuel cells are used.

ENVIRONMENTAL CONTROL AND LIFE SUPPORT SUBSYSTEM

	2-MAN/4-DAY	4-MAN/30-DAY
GAS STORAGE (NO O_2 RECOVERY)	CRYOGENIC	CRYOGENIC
CO_2 REMOVAL	LiOH	2-BED MOLECULAR SIEVE * 4-BED MOLECULAR SIEVE **
WATER MANAGEMENT	STORAGE 97 LB * 192 LB **	MULTIFILTRATION * VACUUM DISTILLATION/PYROLYSIS**
LIFE SUPPORT	16 LB	278 LB
TOTAL WEIGHT	386 LB * 481 LB **	1544 LB * 1724 LB **

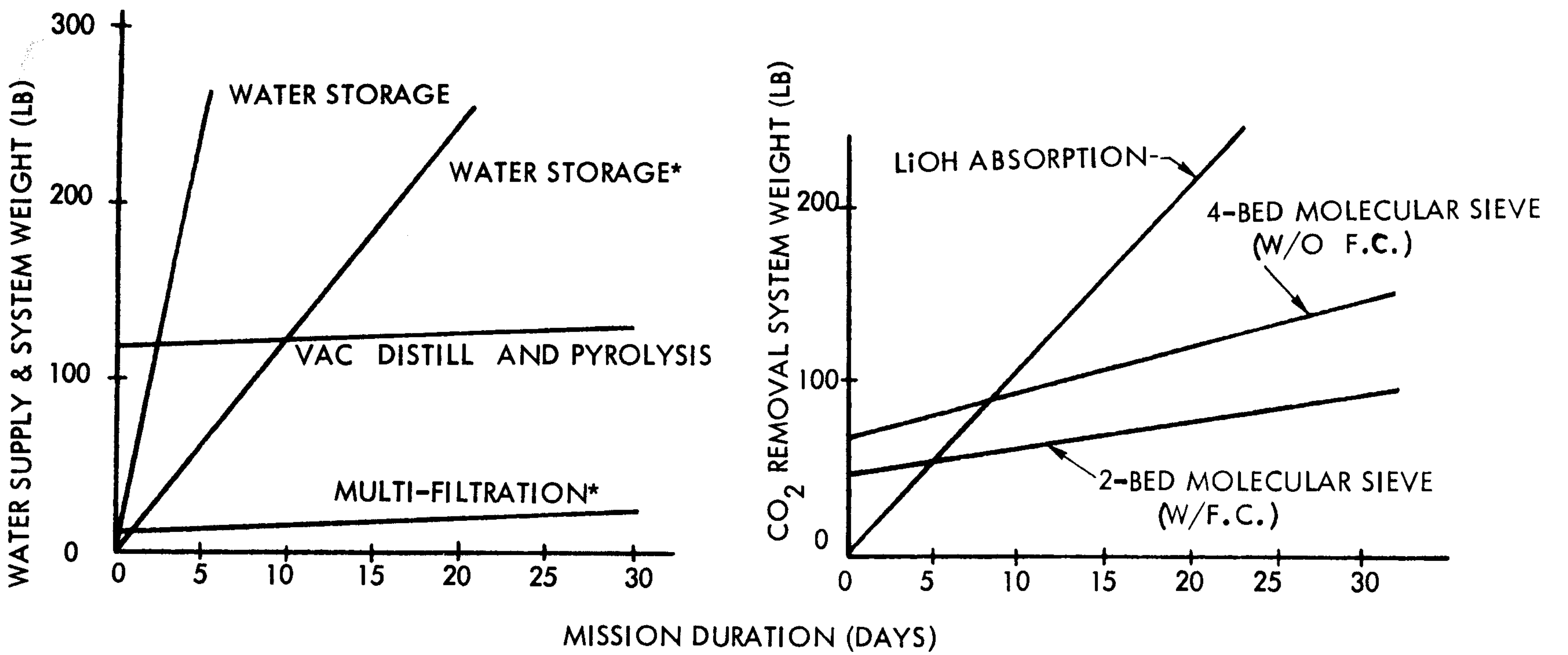
*FUEL CELLS;
**NO FUEL CELLS

WATER MANAGEMENT AND CO_2 REMOVAL SYSTEM

A comparison of water supply system weights indicates that the multifiltration system is lightest for the 4-man/30-day mission when fuel cells are employed. Multifiltration of wash water and condensates is the simplest of the recovery techniques and uses the purest waste water available for recovery. If fuel cells are not used, a vacuum distillation with pyrolysis must be employed to allow the recovery of water from urine.

A two-bed molecular sieve is the lightest CO_2 removal system when fuel cells provide enough water to eliminate the need for water recovery from the silica gel. However, if fuel cells are not used, a four-bed molecular sieve must be employed to permit the recovery of water absorbed by the silica gel.

WATER MANAGEMENT AND CO_2 REMOVAL SYSTEM
4 - MEN



* 2 KW FUEL CELL

GUIDANCE AND CONTROL SUBSYSTEM

The MEM integrated guidance and control system (GCS) provides all guidance, navigation, stabilization, and control information, and functions for the active MEM mission phases (deorbit, entry, retro-propulsion, landing, ascent, and rendezvous). Attitude errors, furnished by the inertial measuring unit and suitably conditioned by the GCS computer, are used to control the reaction control system (RCS) or thrust vector control. The system weighs approximately 190 lb (86 kg).

A two-thrust level RCS is required to provide high roll rates for lift modulation and pitch and yaw damping during entry and low angular rates to minimize propellant consumption during the weightless periods in orbit. Thrust vector control is used to provide attitude control during the powered descent and ascent phases in a manner similar to the lunar module; roll RCS engines provide for roll control. RCS thrust levels and weights are shown for the low L/D MEM.

GUIDANCE AND CONTROL SUBSYSTEM

INTEGRATED G&N AND S&C FUNCTIONS

INERTIAL GUIDANCE SYSTEM (IMU, COMPUTER, ELECTRONICS, OPTICS)
S&C ELECTRONICS
RENDEZVOUS RADAR
RADAR BEACON
LANDING RADAR

SYSTEM WEIGHT

APPROXIMATELY 190 LBS (86 KG)

ATTITUDE CONTROL

UNPOWERED PHASES - ASCENT & DESCENT RCS
POWERED PHASES - THRUST VECTOR CONTROL + ROLL RCS

RCS CHARACTERISTICS

PHASE	THRUST LB	PROPELLANTS LB
ORBIT	185	120
ENTRY	740	2700
ASCENT & RENDEZVOUS	185	120

COMMUNICATIONS SUBSYSTEM

The MEM communications system provides the link between the astronauts, the orbiting spacecraft, and the Earth monitoring stations. The recommended system contains three RF sections (VHF, S-band, and L-band), a television section, and a signal processing section. A MEM/spacecraft/Earth television link was selected because of limitations of MEM antenna size and the high MEM/Earth power requirements.

The VHF communication link provides two-way voice communications between the MEM and spacecraft, and the MEM and astronauts performing EVA. The S-band section provides line-of-sight (LOS) Earth-MEM and MEM-spacecraft communications. The rendezvous radar section consists of a pulse-type radar operating at L-band frequency with an interferometer antenna, in conjunction with a transponder, located in the spacecraft, similar to that used on Gemini. The communications system weight is 587 lb (266 kg) and requires 1045 watts of power.

COMMUNICATIONS SUBSYSTEM

COMMUNICATION LINK	S-BAND (LOS)	VHF (LOS)	L-BAND
MEM/SPACECRAFT	TV-RELAY	TWO-WAY VOICE	RENDEZVOUS RADAR
MEM/EARTH	TWO-WAY VOICE TELEMETRY TRACKING & RANGING	—	—
MEM/EVA	—	TWO-WAY VOICE SUIT DATA	—

WEIGHT - 590 LB
POWER REQ'T - 1050 WATTS

ELECTRICAL POWER SUBSYSTEM

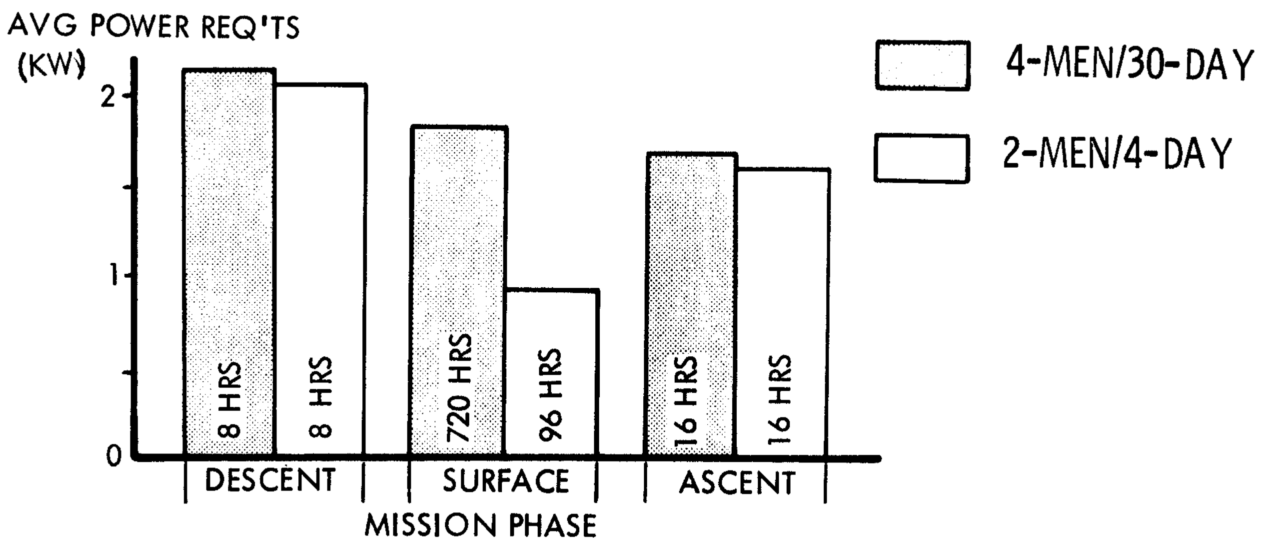
The average subsystem power requirements for these mission phases are shown. Power systems of 1 to 4 KW_e rating were evaluated and their applicability to the MEM mission evaluated on the basis of the power profiles and specific weight characteristics. Fuel cell systems were selected as the primary power source for the descent and surface operations and silver zinc batteries to supplement the descent phase and provide ascent power. Although an isotopic dynamic system was considered for the 4-man 30-day mission, fuel cells were selected because of their advanced state of development and the by-product water produced. The weight of the electrical power subsystem, including distribution and controls, is 2740 lb (1250 kg) for the 4-man 30-day mission and 1295 lb (590 kg) for the 2-man 4-day mission.

ELECTRICAL POWER SUBSYSTEM

SELECTED SYSTEM

	4-MEN/30-DAY	2-MEN/4-DAY
DESCENT & SURFACE OP'NS	2 KW_e FUEL CELL	1 KW_e FUEL CELL*
ASCENT	2 LM ASCENT BATTERIES (16 KWH)	3 LM ASCENT BATTERIES (24 KWH)
WEIGHT (INCL. DISTRIBUTION AND CONTROLS)	2740 LB (1250 KG)	1300 LB (590 KG)

* SUPPLEMENTAL DESCENT POWER (1 KW) PROVIDED BY ASCENT BATTERIES



CREW LIVING QUARTER REQUIREMENTS

The volumes required for crew quarters were investigated briefly. The space available in the Mercury, Gemini, and Apollo varies from 48 to 81 ft^3 (1.4 to 2.3 m^3) of free volume per man. For operations on a planetary surface, various studies indicate that 65 to 90 ft^3 (1.8 to 2.6 m^3) per man are required for two men on a 4-day mission and 120 to 150 ft^3 (3.4 to 4.3 m^3) for four men on a 30-day mission. The total volume shown in the 4-man 30-day MEM designs is approximately 760 ft^3 (21.6 m^3), apportioned in a 2:1 ratio between the laboratory/living quarters and the ascent capsule. Assuming that 20 percent of this volume is for equipment, there remains 150 ft^3 (4.3 m^3) of free volume per man.

CREW LIVING QUARTER REQUIREMENTS

SOURCE	CREW SIZE	MISSION DURAT'N (DAYS)	TOTAL VOLUME (FT^3)	FREE VOL/MAN (FT^3)
MERCURY	1	2	106	48
GEMINI	2	14	175	53
APOLLO				
CM	3	15	305	75
LM	2	2	235	81
STUDIES	2	4	130-180	65-90
	4	30	480-600	120-150
MEM DESIGN			(LAB + ASCENT CAPSULE)	
LOW L/D	4	30	767	154
LIFTING BODY	4	30	755	151

WEIGHT COMPARISONS

Comparative weights are shown for the low and moderate L/D configurations. Weights for the circular orbit missions range from 66,800 pounds (30,400 kg) for the low L/D 2-man/4-day configuration which employs ballute/retro retardation to 91,800 pounds (41,700 kg) for a 4-man/30-day lifting-body design which uses retropropulsive descent; the recommended low L/D configuration for the selected 4-man/30-day design mission weighs 109,000 pounds (49,500 kg). The low L/D design is 3 to 12 percent lighter than the lifting body (depending on the mission parameters and retardation concept) because of the reduced retardation ΔV required; ballute/retroretardation systems are lighter than all-retropropulsive descent. Reducing the crew from four to two men leads to a 10 percent reduction in weight; a reduction in stay time from 30 to 4 days, however, has only a small effect on weight.

WEIGHT COMPARISONS
(LB)

		LOW L/D				LIFTING BODY	
		ELLIPTICAL ORBIT (ΔV = 20350 FPS)		CIRCULAR ORBIT (ΔV = 16000 FPS)		CIRCULAR ORBIT (ΔV = 16000 FPS)	
		2-MAN	4-MAN	2-MAN	4-MAN	2-MAN	4-MAN
RETRO PROPULSIVE DESCENT	4 DAY	88,500	104,500	70,600	76,300	79,200	86,800
	30 DAY	91,700	109,000	73,800	80,700	82,800	91,800
BALLUTES/ PROPULSIVE RETARDATION	4 DAYS	84,500	97,300	66,800	72,100	69,200	74,100
	30 DAYS	87,600	102,00	69,900	75,500	72,600	79,000

BASIS:
2 MINUTE HOVER

WEIGHT SENSITIVITIES

The MEM weight sensitivities to propellants, staging (engines and living quarters) and hover time are shown. Cryogenic propellants such as fluorine/hydrogen result in conceptually lighter MEM's but suffer from volumetric deficiencies. Earth storables on the other hand lead to heavier but more compact designs.

One stage for descent and two for ascent result in minimum weight; only one engine was selected for ascent (1-1/2 stages) because it imposes less packaging problems. The use of separate living quarters (which are left behind on Mars) also appears to be attractive, even for the 2-man 4-day mission. Hover time affects the MEM weight to the extent of approximately 9 percent for each minute of capability.

WEIGHT SENSITIVITIES
(LB)

		LOW L/D CONFIGURATION		LIFTING BODY
		ELLIPTICAL ORBIT	CIRCULAR ORBIT	CIRCULAR ORBIT
PROPELLANTS	I_{SP}			
SPACE STORABLES	383 (SEC)	109,000	80,700	91,800
CRYOGENICS	456	82,800	67,600	83,900
EARTH STORABLES	338	141,900	102,900	122,900
EFFECT OF STAGING				
SINGLE STAGE DESCENT/ASCENT		188,000	139,300	158,400
SINGLE STAGE DESCENT/ONE STAGE ASCENT		113,600	84,200	95,600
1-1/2 STAGE ASCENT		109,000	80,700	91,800
2 STAGE ASCENT		108,100	80,000	91,000
EFFECT OF ASCENT CAPSULE/ LIVING QUARTERS				
2 MEN/4 DAYS				
SINGLE CABIN		98,500	76,600	85,800
SEPARATE LIVING QUARTERS		88,500	70,600	79,200
HOVER TIME				
0		89,800	66,600	76,500
1 MIN		98,900	73,300	83,300
2 MIN		109,000	79,500	91,400

BASIS: 4-MEN/30-DAYS MISSION
RETRO PROPULSIVE DESCENT
2 MINUTES HOVER

MEM DESIGN—LOW L/D

A low L/D 30-foot (9.1 m) diameter Apollo-shape configuration was selected as the recommended design. This MEM would be capable of performing the maximum mission, i.e., landing 4 men for 30 days and returning them to a 162 by 36,100 nm (300 by 66,900 km) orbit (e = 0.9).

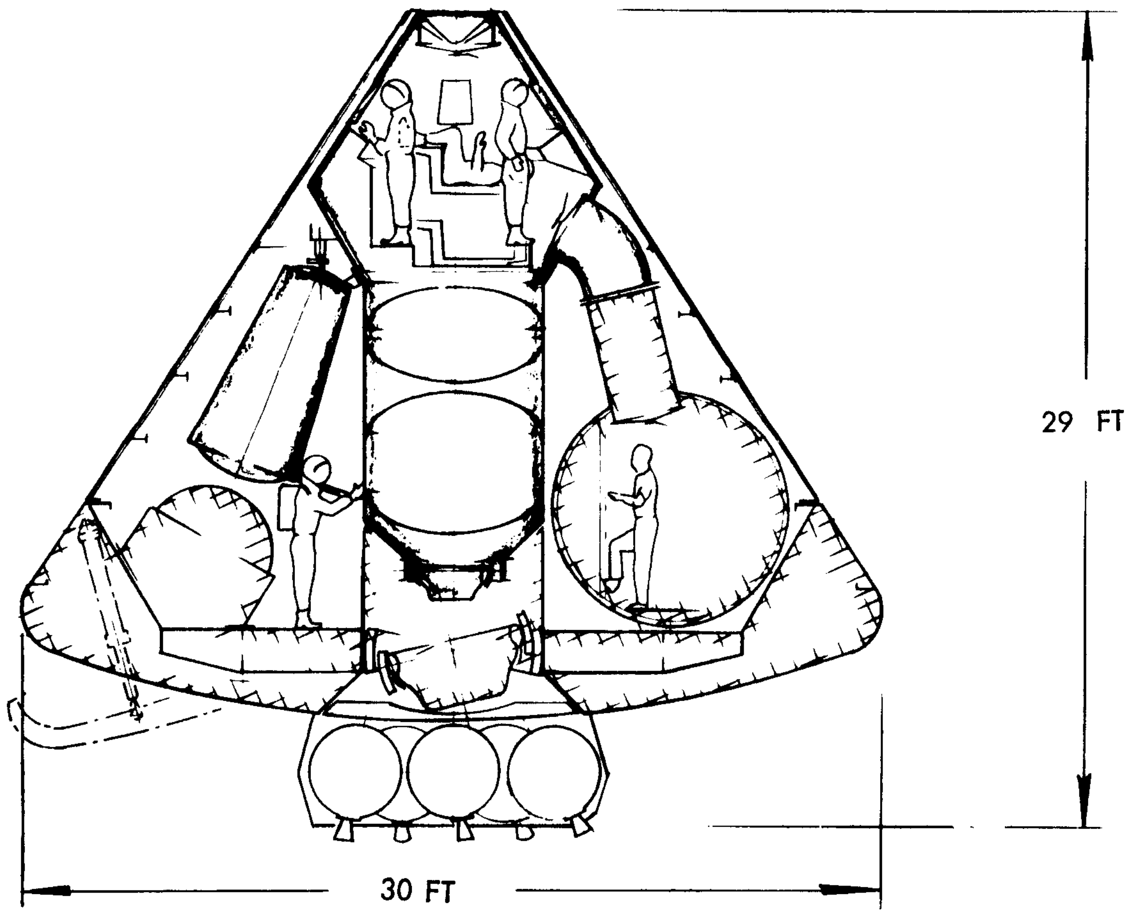
The deorbit motors are jettisoned after firing and the vehicle enters the atmosphere at an angle of attack of 147 degrees. The conical section of the heat shield and the plug over the retro-engine are jettisoned prior to igniting the retropropulsion engine. Although the crew compartment contains four couches, two of the crew will assume standing positions to land the vehicle. A six-legged landing gear, which is an integral part of the aft heat shield, attenuates the landing loads. (The landed stage of the vehicle is shown by the cross-hatching and the ascent stage by the shading.)

The crew quarters and laboratory are in the form of a toroidal section and are connected to the crew capsule by an airlock. Another airlock provides access to the Martian surface and interior of the vehicle for inspection and maintenance.

A 140,000-pound-(64,000 kg) thrust plug nozzle descent engine and a 35,000-pound (16,000 kg) ascent engine are arranged symmetrically. Both engines are gimballed; the descent engine is canted approximately 13 degrees to thrust through the off-set c.g. The c.g. off-set is achieved by arranging the eight spherical descent propellant tanks and eight first-stage ascent tanks assymetrically around the central structure. The two second-stage ascent tanks are mounted on the central thrust structure between the crew cabin and the ascent engine.

The vehicle weight would be 109,000 pounds (49,400 kg) at separation; the m/C_LA at entry of 7.4 slugs/ft^2 (1160 kg/m^2) is similar to that of the Apollo command module. To allow for possible growth in the vehicle or payload weight, a larger vehicle may be desirable. For example, given the same general arrangement and employing a 31.5 foot (9.6 m) base diameter, the gross weights can increase by approximately 50 percent and the highly elliptical orbit missions can still be accomplished.

MEM DESIGN
LOW L/D



CHARACTERISTICS

4 MAN/30 DAY

W_{GROSS} = 109,000 LBS

$M/C_L A$ = 7.4 SLUGS/FT^2

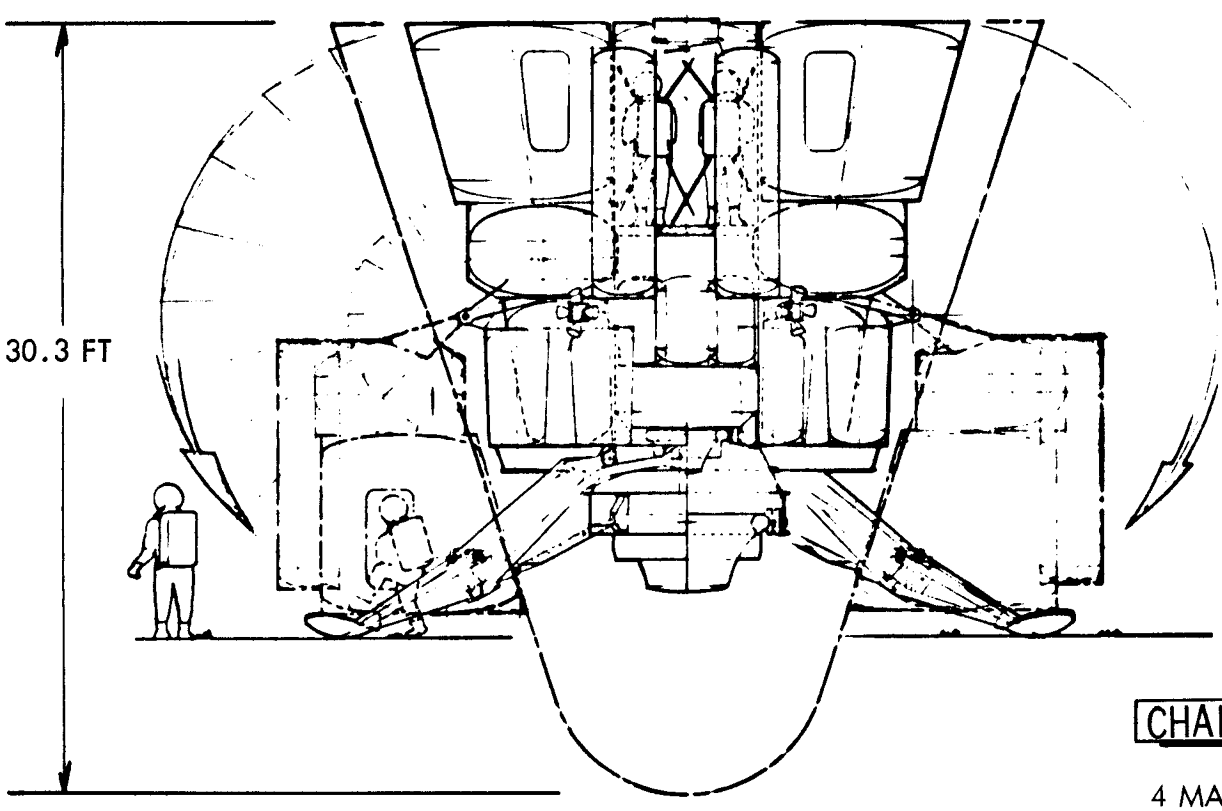
ASCENT ΔV ($e = 0.9$) = 20,350 FPS

MEM DESIGN—LIFTING BODY

A lifting-body MEM design is shown. The configuration consists of a blunted half cone—an 18-degree half-cone angle—which flies with the flat surface down at an angle of attack of 40 degrees. During deorbit, entry, and ascent, the 4-man crew is stationed in a minimum size capsule at the top of the vehicle. In the design shown, the crew is supported in a standing position by harnesses with lightweight back supports to take the low eyeballs-in accelerations during entry. The external thermo-structural shell is jettisoned after entry to allow for deployment of the landing gear and use of the 100,000 pound (45,000-kg) descent engine at the front of the vehicle. Landing occurs in a nose-down attitude as shown. The crew quarters and a laboratory, positioned at the rear of the vehicle before landing, are deployed as shown. The living quarters, landing gear and descent stage are left behind on the surface. The ascent stage consists of the ascent capsule, 30,000-pound (13,600-kg) thrust ascent engine, and the ascent tanks, which are staged.

The gross weight of this design for the low circular orbit mission is 83,000 pounds (38,000 kg). The weight will increase to 120,000-pounds (54,600 kg) for the elliptical orbit mission (e = 0.9); the length dimension would have to be increased to approximately 37 ft (11.6 m) to accommodate the additional propellants.

MEM DESIGN
LIFTING BODY



CHARACTERISTICS

4 MAN/30 DAY
W_{GROSS} = 86000 LB
$M/C_{L}A$ = 8.1 SLUGS/FT^2
ASCENT ΔV = 16000 FPS

DETAILED WEIGHT STATEMENT

ASCENT STAGE

Detailed weight statements are presented for:

a. The recommended 4-man 30-day, low L/D 30 ft (9.1 m) diameter design configured for the elliptical orbit (e = 0.9) mission.

b. A minimum size, 4-man 30-day, low L/D 26.5-ft (8.1-m) diameter, configured design for the low circular orbit mission.

c. A minimum size, 4-man 30-day lifting body design which is 30.3 ft (9.2 m) long, and is configured for the low circular orbit mission.

All three designs incorporate the recommended retropropulsive descent system and a heat shield designed for entry from Earth orbit. No weight growth is shown in these statements.

The ascent capsule which serves as the control center and houses the crew during both entry and ascent weighs 5260 lb (2386 kg) for each of the three vehicles; the total ascent stage weighs the same for the low L/D and lifting-body designs for the low circular orbit mission.

DETAILED WEIGHT STATEMENT *

ASCENT STAGE

	LOW L/D D=30FT(9.1M) 20.3KFPS(6.2KPS) ELLIPTICAL ORBIT		LOW L/D D=26.5FT(8.1M) 16KFPS(4.9KPS) LOW CIRCULAR ORBIT		LIFTING BODY L=30.3FT(9.2M) 16KFPS(4.9KPS) LOW CIRCULAR ORBIT	
	LB	KG	LB	KG	LB	KG
ASCENT CAPSULE	(5260)	(2386)	(5260)	(2386)	(5260)	(2386)
STRUCTURE	980	445	980	445	980	445
POWER	500	227	500	227	500	227
COMMUNICATIONS	210	95	210	95	210	95
GUIDANCE & CONTROL	225	102	225	102	225	102
ECLSS	1340	608	1340	608	1340	608
RCS	530	240	530	240	530	240
RETURN PAYLOAD	300	136	300	136	300	136
CREW (90 PERCENTILE)	700	318	700	318	700	318
CONTINGENCY	475	215	475	215	475	215
STAGE II	(9430)	(4277)	(6510)	(2953)	(6510)	(2939)
TANKS & SYSTEM	690	313	470	213	470	213
ENGINE	490	222	300	136	300	136
PROPELLANT	8250	3742	5740	2604	5740	2590
STAGE I	(22,510)	(10,210)	(12,830)	(5820)	(12,830)	(5820)
TANKS & SYSTEM	1610	730	910	413	910	413
PROPELLANT	20900	9480	11920	5407	11920	5407
TOTAL WEIGHT	37,200	16,873	24,600	11,159	24,600	11,145

* 4-MAN/30-DAY MISSION

DETAILED WEIGHT STATEMENT

DESCENT STAGE

The detailed weight statement is continued to show the entry weight and gross weight (i. e. before deorbit) of the three designs. The low L/D design reflects jettisoning of the major portion of the conical heat shield and structure before landing, while on the lifting body only the nose portion is jettisoned. Other subsystem payloads and propellant weights are in accordance with the requirements analyses.

The gross weight of the low L/D design is 109,000 lb (49,400 kg) for the elliptical orbit mission and 73,300 lb (33,200 kg) for the low circular orbit mission. This difference directly reflects the difference in ascent capsule weights. The corresponding weight for the lifting body for the low circular orbit mission is 83,300 lb (37,800 kg), i. e., 10,000 lb (4550 kg) heavier than the low L/D design. Less than 1000 lb (455 kg) of this difference is the result of jettisoning the smaller structural weight before retropropulsion, 2250 lb (1000 kg) is due to the more complex laboratory design in the lifting body vehicle; and the balance of 6000 lb (2700 kg) is due to the greater descent ΔV required for the lifting-body configuration. The solid deorbit motors represent about 7 percent of the total MEM weight.

DETAILED WEIGHT STATEMENT

DESCENT STAGE

	LOW L/D D=30FT(9.1FT) 20.3KFPS(6.2KPS) ELLIPTICAL ORBIT		LOW L/D D=26.5FT(8.1M) 16KFPS(4.9KPS) LOW CIRCULAR ORBIT		LIFTING BODY L=30.3FT(9.2M) 16KFPS(4.9KPS) LOW CIRCULAR ORBIT	
	LB	KG	LB	KG	LB	KG
ASCENT STAGE	(37,200)	(16,873)	(24,600)	(11,158)	(24,600)	(11,158)
DESCENT STAGE						
JETTISONED STRUCTURE	4,650	2,109	3,500	1,588	1,030	467
RETAINED STRUCTURE	6,350	2,880	5,000	2,268	6,170	2,799
LABORATORY STRUCTURE	1,350	612	1,350	612	3,600	1,633
EPS	2,240	1,016	2,240	1,016	2,240	1,016
COMMUNICATIONS	370	168	370	168	370	168
GUIDANCE & CONTROL	10	5	10	5	10	5
ECLSS	1,630	739	1,630	739	1,630	739
RCS	2,630	1,193	1,590	721	1,990	903
LANDING GEAR	2,770	1,256	2,020	916	2,120	962
NET LANDED PAYLOAD	4,200	1,905	4,200	1,905	4,200	1,905
CONTINGENCY	3,070	1,393	2,520	1,143	2,750	1,247
TANKS & PROPULSION	2,600	1,179	1,410	640	2,000	907
ENGINE	2,020	916	1,350	612	1,540	699
PROPELLANT	30,500	13,835	16,510	7,488	23,400	10,614
ENTRY WEIGHT	101,590	46,079	68,300	30,980	77,650	35,222
DEORBIT MOTORS	7,400	3,357	5,000	2,268	5,650	2,570
TOTAL WEIGHT	108,990	49,436	73,300	33,248	83,300	37,792

DESIGN COMPARISONS

A comparison of the low L/D and lifting-body configurations shows that the lifting body is 10 percent heavier for retropropulsive descent. The 4-man 30-day elliptical orbit mission design results in 45 percent more weight than the low circular orbit design.

The growth potential can be expressed in terms of the maximum ascent ΔV capability of the MEM since the packaging of the ascent propellant is limited by the dimensions of the vehicle. A 31.5-ft (9.6-m) diameter size constraint has been assumed based on a concurrent study being conducted by NAR for NASA on technology requirements for aerobraker spacecraft (NAS2-4135). Within this size constraint, the low L/D design can achieve 22,300 fps (6.8 km/sec) and the lifting body 21,000 fps (6.4 km/sec). The requirements for the low circular and elliptical orbit missions are 16,000 fps (4.9 km/sec) and 20,350 fps (6.2 km/sec). Both designs can perform all missions considered; however, the low L/D configurations shows more growth potential. Both configurations appear compatible with the packaging, center of gravity, and operational requirements of the aerobraker spacecraft.

DESIGN COMPARISONS

(4 MEN/30 DAY MISSION)

	LOW L/D	LIFTING BODY
WEIGHTS (LB)		
LOW CIRCULAR ORBIT	65,000 - 75,000	67,000 - 85,000
ELLIPTICAL ORBIT	109,000	120,000
GROWTH POTENTIAL (31.5 FT DIAM)		
MAX. ASCENT ΔV (FPS)	22,300	21,000
MISSION PERFORMANCE	BOTH CONFIGURATIONS ACCEPTABLE	
SPACECRAFT INTERFACE A/B OR R/B		

MEM DESIGN SUMMARY

A low L/D or lifting-body configuration can perform the full range of missions defined from low circular to highly elliptical orbits and still be compatible with a 33-ft (10-m) diameter trans-Mars spacecraft. Weight growth during development of up to 50 percent can be accommodated in the low L/D design; the growth potential of the lifting body is significantly lower. The weight of the low L/D design will be 10 percent lighter than the corresponding lifting-body design.

The low L/D design is recommended as the preferred MEM configuration because of its lower weight, its greater growth potential within the size constraint, and the more mature aero-thermo-structural technology, which will be available in the time frame of interest as compared to a lifting-body configuration. The MEM test program, which follows, must be compatible with either design to allow for future flexibility. Since the subsystems, critical development issues, and operational procedures are essentially identical for the two designs, this recommendation has been considered in developing the recommended test program.

MEM DESIGN SUMMARY

CONCLUSIONS

BOTH CONFIGURATIONS

CAN PERFORM ELLIPTICAL ORBIT MISSION
COMPATIBLE WITH 33 FT (10 M) DIA SPACECRAFT

LOW L/D CONFIGURATION

10% LIGHTER THAN LIFTING BODY
MATURE ENTRY TECHNOLOGY
ORBIT MISSION

RECOMMENDATIONS

LOW L/D DESIGN PREFERRED
TEST PROGRAM SHOULD BE COMPATIBLE WITH EITHER DESIGN

MEM TEST PROGRAM

BRIEFING OUTLINE

The MEM test program begins with a brief description of the test philosophy and test requirements. Candidate tests classified as ground tests, flight tests (both suborbital and orbital), and unmanned Mars flight tests are described. Applicable scaling requirements for each test have been analyzed, and the results are presented. The resources analyses considered schedules, facilities, and costs of the recommended test program. Cost tradeoffs on alternative MEM designs and test programs were conducted which evolved in a recommended MEM development program.

MEM TEST PROGRAM
BRIEFING OUTLINE

BACKGROUND AND SCOPE	
CANDIDATE TESTS	GROUND TESTS FLIGHT TESTS UNMANNED MARS FLIGHT TESTS
RESOURCES REQUIREMENTS	SCHEDULES FACILITIES COST AND COST TRADE-OFFS
RECOMMENDED PROGRAM	

MANNED SPACECRAFT DEVELOPMENT

SUMMARY

Test and development experience gained from previous manned space programs indicates that our understanding of the effects of the space environment (including weightlessness) on the physiological and operational capabilities of the astronauts follows an evolutionary process. The Mercury program demonstrated man's ability to endure rigors of space flight. The Gemini program further demonstrated man's ability to perform useful tasks in space including EVA. The Apollo program will verify the ability of man to accomplish lunar exploration - the threshold of interplanetary travel.

Hardware testing on the other hand generally begins on the component level for each new spacecraft program with qualification tests and is consumated with the manned flights. Where components or subsystems have been developed on a previous program (e.g., recovery aids on Apollo), testing may begin at some higher level of qualification.

MANNED SPACECRAFT DEVELOPMENT

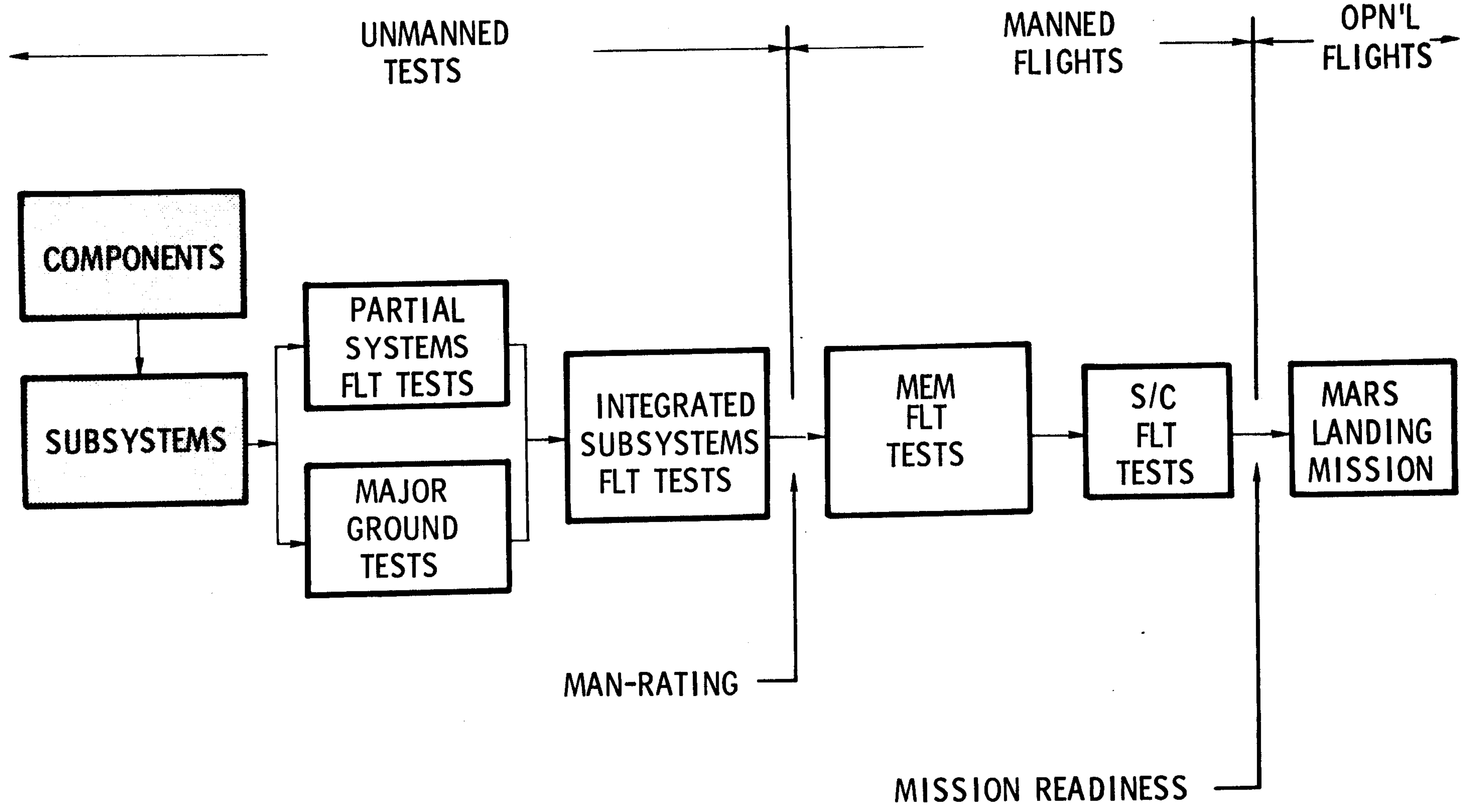
- SUMMARY -

- ASTRONAUT PERFORMANCE HAS INCREASED WITH INCREASED MISSION SOPHISTICATION
- NEW SYSTEM EQUIPMENTS TESTING BEGINS AT LOWEST LEVEL THROUGH INTEGRATED SYSTEMS TESTS

TEST PROGRAM SCOPE

The total test program suggested for the MEM will start at the component level, progress through subsystems and progressively higher levels of testing, and culminate with the manned Mars landing mission. Testing at the component and subsystem level was not considered in depth in this study. The major ground tests (i. e., those which require partial or complete test vehicles), partial systems flight tests, unmanned and manned MEM flight tests, and integrated systems spacecraft flights (as they apply to the MEM) which precede the Mars landing mission were investigated.

TEST PROGRAM SCOPE



TEST REQUIREMENTS

The various operational phases of the MEM, from acceptance of the vehicle through the docking at Mars just prior to MEM abandonment, are indicated. The column on the left lists the MEM subsystems, including structure and aerodynamic configuration. The matrix of open and solid circles indicates when the various subsystems are active; the dashed lines indicate the operational environment in which subsystems must survive in a dormant mode before becoming active. The typical mission timeline indicates the durations of the respective phases.

The solid circles identify the major development issues which require either an advance in the state-of-the-art (such as the descent and ascent propulsion subsystem) or involve a new application of current technology (such as use of the landing gear on Mars terrain).

The test requirements indicate that each subsystem should be tested under the conditions and environments which would exist when the subsystem is active (shown by the circles) and after the subsystem has been dormant for the appropriate length under environments similar to those shown by the dashed lines. For example, the descent reaction control system (RCS) must be acceptance tested; exposed to atmospheric conditions representing pre-launch operations for 30 days; boost levels of vibrations and loads; Earth orbital, space and Mars orbit vacuum and temperature conditions for 242 days; and then activated and tested under entry loads and temperatures. Simulation may be used where necessary to duplicate the loads, temperatures, and other required environments.

TEST REQUIREMENTS

OPERATIONAL PHASE

SUBSYSTEM	ACCEPTANCE	PRELAUNCH	BOOST	E.O. OPERATIONS	MARS INJECTION	TRANSIT	MARS CAPTURE	ORBITAL CHECKOUT	SEPARATION	DEORBIT	ENTRY	TERMINAL DESCENT	LANDING	MARS SURF. OPERATIONS	PRELAUNCH CHECKOUT	ASCENT	RENDEZVOUS	DOCKING
RCS - DESCENT	0	- - -	- - -	- - -	- - -	- - -	- - -	- - -	- - -	- - -	●							
ORBITAL	0	- - -	- - -	- - -	- - -	- - -	- - -	- - -	●	0	- - -	- - -	- - -	- - -	- - -	- - -	●	0
PROPULSION - RETRO	- - -	- - -	- - -	- - -	- - -	- - -	- - -	- - -	- - -	●								
- DESCENT	0	- - -	- - -	- - -	- - -	- - -	- - -	- - -	- - -	- - -	- - -	●						
- ASCENT	0	- - -	- - -	- - -	- - -	- - -	- - -	- - -	- - -	- - -	- - -	- - -	- - -	- - -	- - -	●		
GUIDANCE AND CONTROL	0	- - -	- - -	- - -	- - -	- - -	- - -	0	0	0	●	0	●	- - -	0	0	0	0
THERMAL PROTECT - H/S	- - -	- - -	- - -	- - -	- - -	- - -	- - -	- - -	- - -	- - -	●							
- INSUL	- - -	0	0	0	0	●	0	0	0	0	●	0	0	●	0	0	0	0
POWER SYSTEM	0	- - -	- - -	- - -	- - -	- - -	- - -	0	0	0	0	0	0	●	0	0	0	0
ECLSS	0	- - -	- - -	- - -	- - -	- - -	- - -	0	0	0	0	0	0	●	0	0	0	0
COMMUNICATIONS	0	- - -	- - -	- - -	- - -	- - -	- - -	0	0	0	●	0	0	0	0	0	0	0
BALLUTES/CHUTES	- - -	- - -	- - -	- - -	- - -	- - -	- - -	- - -	- - -	- - -	- - -	●						
MECHANISMS - DOCKING	- - -	- - -	- - -	- - -	- - -	- - -	- - -	- - -	- - -	- - -	- - -	- - -	- - -	- - -	- - -	- - -	- - -	0
- LANDING GEAR	- - -	- - -	- - -	- - -	- - -	- - -	- - -	- - -	- - -	- - -	- - -	- - -	●					
- SEP DEVICE	- - -	- - -	- - -	- - -	- - -	- - -	- - -	- - -	0	●	- - -	●	●	- - -	- - -	●		
STRUCTURE	- - -	- - -	●	- - -	0	- - -	●	- - -	- - -	- - -	●	●	●	- - -	- - -	●	- - -	0
AERODYNAMIC CONFIG											●	●						
CHECKOUT SENSORS	0	- - -	- - -	0	- - -	- - -	- - -	0	- - -	- - -	- - -	- - -	- - -	- - -	0			
TYPICAL TIMELINE (DAYS)	-90	-30	0	40	40	240	240	241	242	242	242	242	242	271	272	272	272	272

- - - DORMANT
0 ACTIVE
● MAJOR DEV'T ISSUES

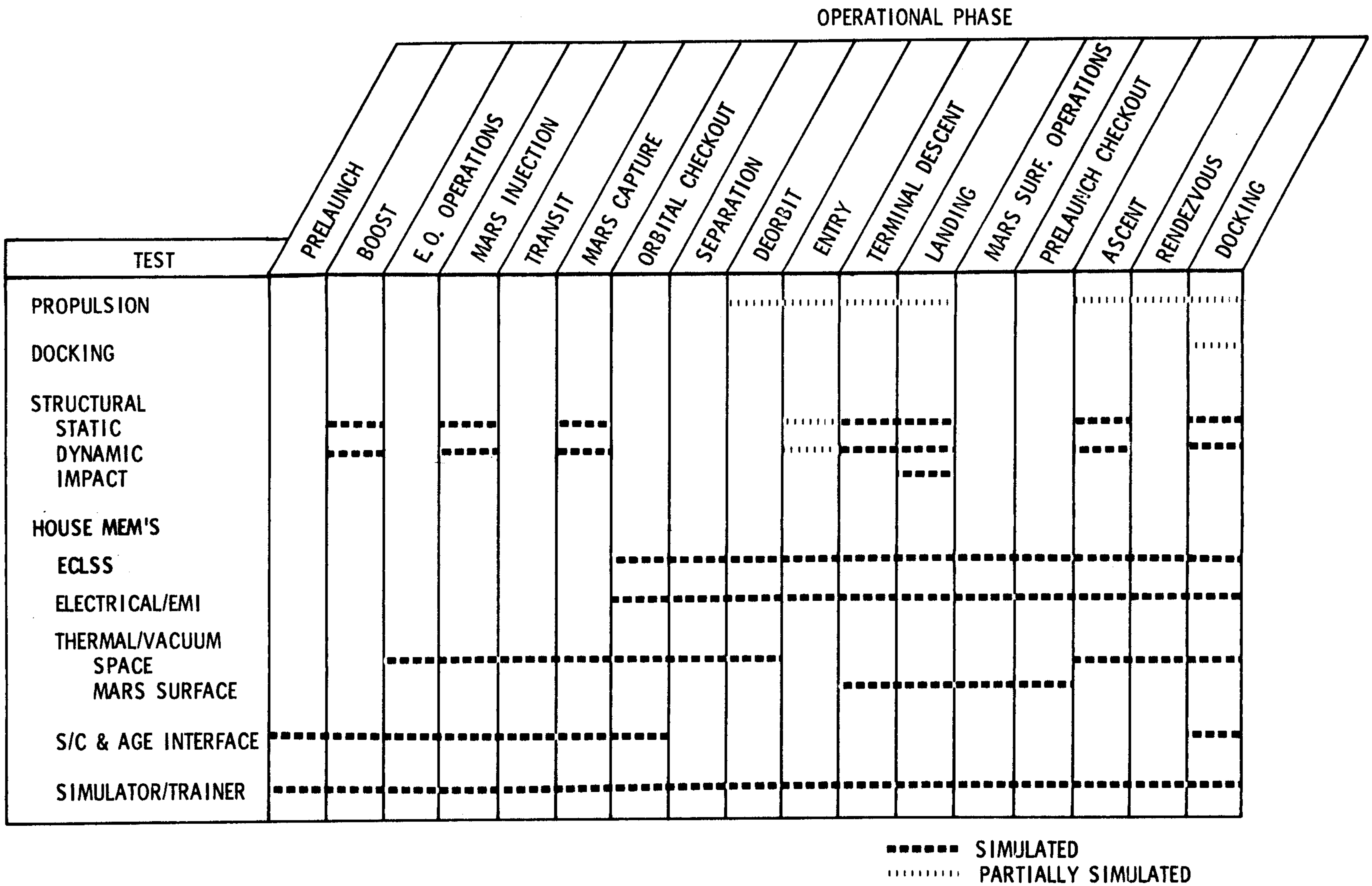
CANDIDATE GROUND TESTS

SUMMARY

Major ground tests (i. e., those that require a partial or complete MEM test vehicle) and their respective operational phases are identified. The propulsion, docking, and structural tests are directed toward developing and verifying the performance of specific subsystems; the other tests are integrated subsystems tests and will be conducted on house MEM's. For example, the thermal/vacuum testing will be conducted on all the subsystems performing their functions insofar as is possible, within a space chamber under simulated conditions of temperature, pressure, etc.

The degree to which the MEM environment can be simulated will vary. Similitude will be assumed when the relevant environmental parameters are produced for a test, although not necessarily by the same combination of physical parameters as on the MEM mission. When an important element of the environment, or of the hardware, is missing the simulation is only partial, and the test does not resolve all the issues. It appears that simulation can be achieved for all the tests except for the propulsion and structural tests during Mars entry.

CANDIDATE GROUND TESTS
SUMMARY

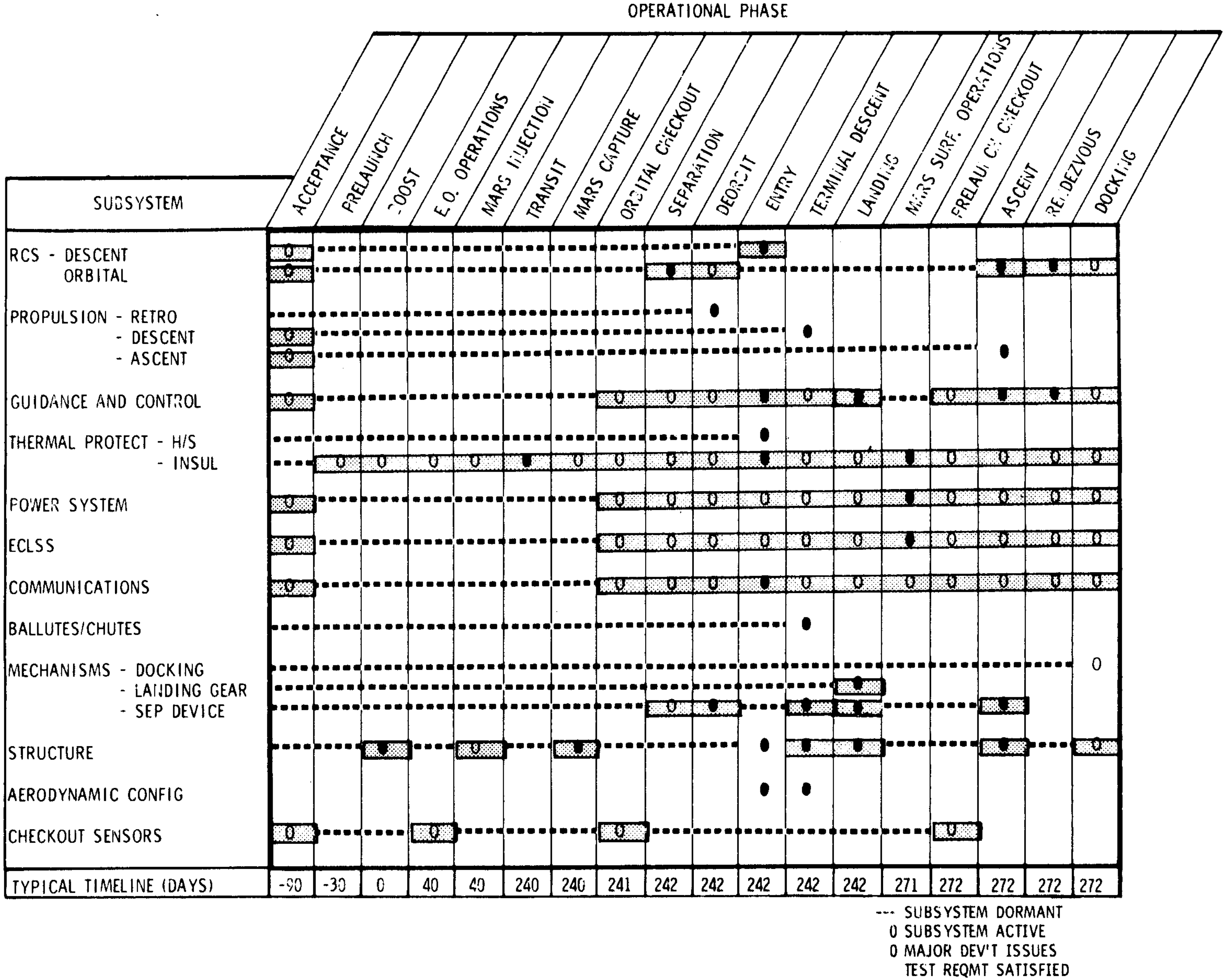


SD 67-755-4

TEST REQUIREMENTS SATISFIED BY GROUND TESTS

A large number of the test requirements identified can be satisfied by the candidate ground tests. For example, the RCS, guidance and control, insulation, power, ECLSS, and communications systems can be fully tested in ground tests. The structure on the other hand cannot be adequately ground tested for the entry phase since the transient temperature environment cannot be reproduced during either static or dynamic testing. The requirement for full duration tests of the retro and ascent propulsion systems in vacuum and the descent system in high-speed atmospheric flight cannot be fully satisfied in ground tests, although firings in vacuum chamber facilities such as those at AEDC can simulate Mars altitudes to above 200,000 ft (61 km). It is also obvious that full-scale testing of the heat shield, ballutes (if required), and the aerodynamic configuration require flight testing. These tests can be conducted either on an individual subsystem level or incorporated into an integrated systems flight test.

TEST REQUIREMENTS SATISFIED BY GROUND TESTS



CANDIDATE EARTH/CISLUNAR FLIGHTS

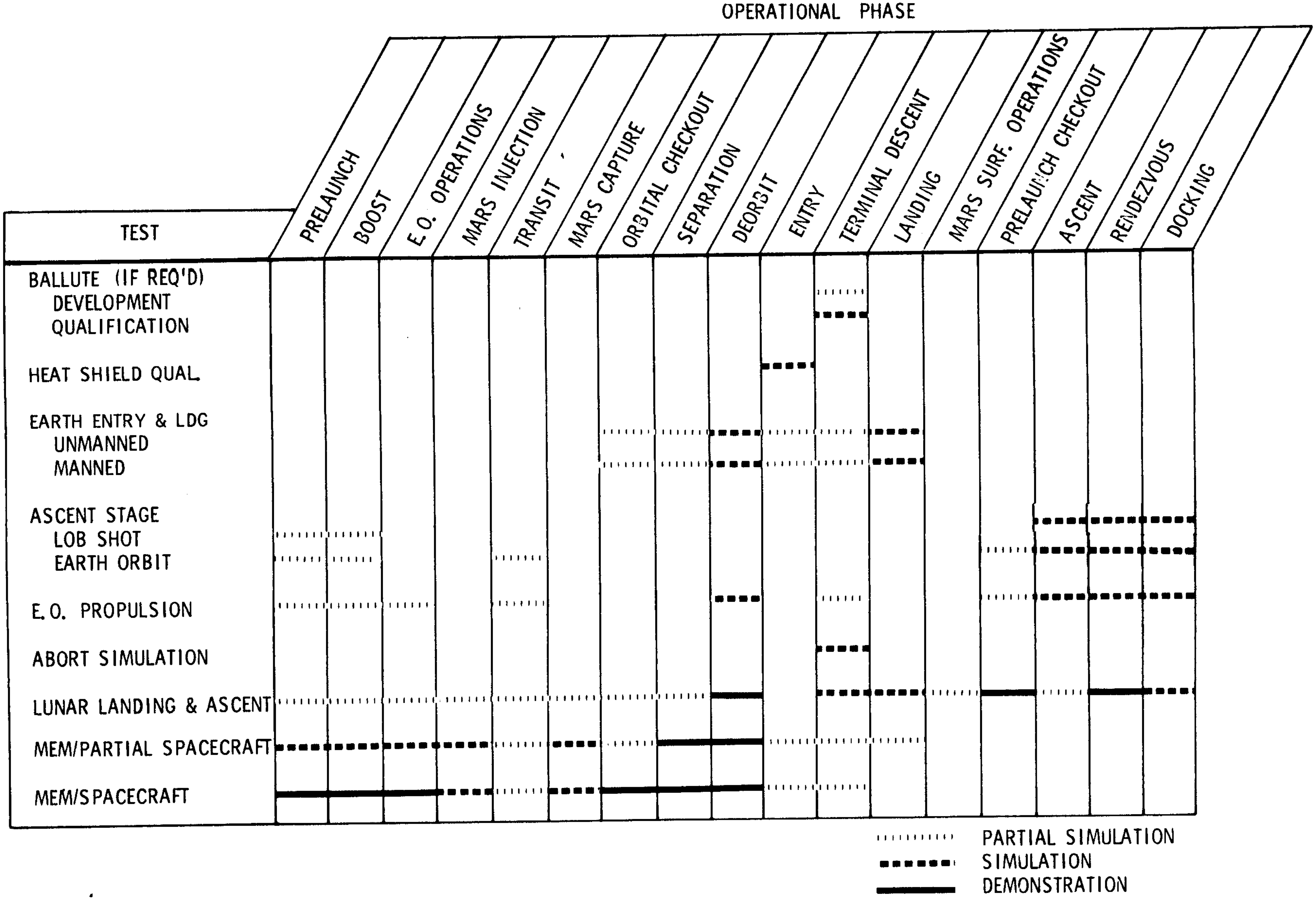
SUMMARY

A number of candidate flight tests have been identified and evaluated. These include tests for developing and qualifying specific subsystems, such as the ballutes (if required) and the heat shield; Earth orbital and atmospheric flights to test the propulsion subsystems, verify vehicle aerodynamics, and provide crew flight experience; and lunar and cislunar flights of integrated MEM trans-Mars spacecraft and MEM subsystems to enhance confidence in the capability to perform the Mars landing mission. A flight abort test also has been considered.

Full environmental simulation can be obtained of the critical mission phases and partial simulation of other phases. For example, in the Earth entry and landing tests orbital checkout and separation may be carried out in a shroud enclosing the MEM test vehicle rather than the interplanetary spacecraft. The ballute, heat shield qualification, Earth entry and landing, and abort flights simulate mission phases which are not repeated by other near-Earth flight tests. However, the phases covered by the two candidate ascent stage flights are duplicated in the Earth orbital propulsion test.

The last three tests provide increasingly greater degrees of simulation and demonstration (i. e., in some phases the environment is essentially duplicated rather than simulated). A fully integrated MEM/spacecraft flight can provide a meaningful demonstration of the MEM mission phases which occur from Earth orbital operations to deorbit.

CANDIDATE EARTH/CIS-LUNAR FLIGHTS SUMMARY



SD 67-755-4

HEAT SHIELD QUALIFICATION TEST

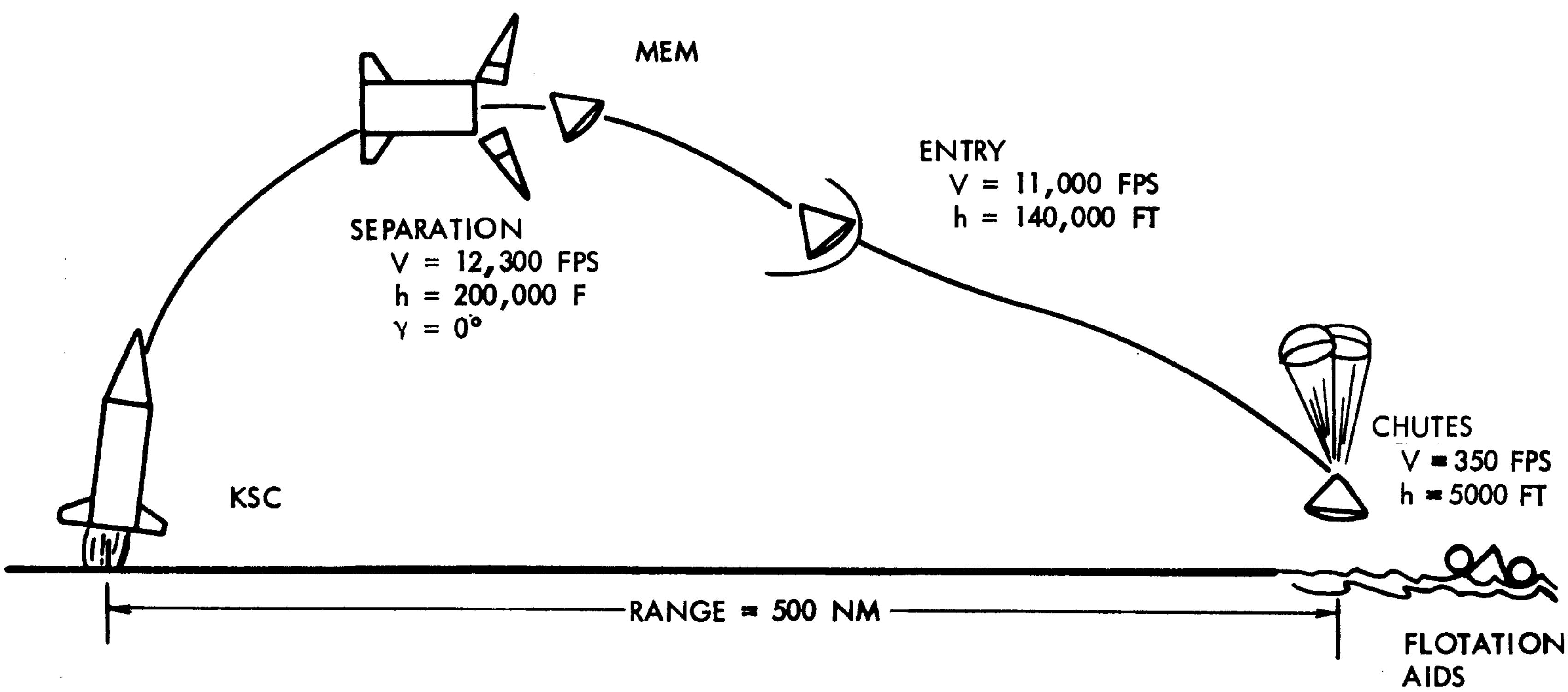
The heat shield can be qualified in suborbital flight tests within the Earth's atmosphere, which duplicate the Mars design entry heating pulse at approximately the same velocity and density as at Mars. Since the structure and insulation are part of the heat protection system, a MEM vehicle must be used, but only the subsystems shown are essential to this test. Scaling analyses indicated that the mass of the test vehicle must be 75 to 125 percent of the MEM mass, depending on the Mars atmosphere design model.

Launch would be from KSC on an off-loaded Saturn V (to match the MEM diameter); recovery would occur at sea approximately 500 nm (900 km) downrange. Since the MEM retardation and propulsion subsystems may not have been qualified at the time of this test, a cargo parachute subsystem and a water impact and recovery subsystem would be required.

The possibility of designing the MEM heat shield for Earth entry would eliminate the need for this suborbital test, but would result in additional weight to the MEM. However, the heat shield can then be qualified in the unmanned entry and landing test from Earth orbit.

HEAT SHIELD QUALIFICATION TEST

PARTIAL MEM SUBSYSTEMS
- STRUCTURE
- THERMAL PROTECTION
- SEPARATION DEVICES
- S&C
- RCS



HEATING RATE SCALING

To test the MEM heat shield in the Earth's atmosphere, a trajectory that matches the heating rate-time history for the Mars design conditions must be flown. Mars entry at a speed of 11,050 fps (3.4 km/sec), an entry angle of -7 degrees, and an m/C_LA of 6.4 slugs/ft^2 (1000 kg/m^2) was considered. This represents the maximum heating rate design condition for the low circular orbit MEM mission, i.e., entry from a 270-nm (500-km), circular parking orbit. The required Earth trajectory is obtained by (1) matching the ambient density at Mars peak heating, (2) choosing a velocity 7 percent greater than for Mars to allow for possible differences in transport properties, and (3) using the equation shown to obtain a local flight path angle and assure that, at these selected conditions of the trajectory, the heating rate is at a maximum. The value of m/C_LA is selected so that the duration of the heating pulse can be adjusted as required.

The two figures illustrate the match of Mars and Earth entry heating pulses using these scaling laws. For the VM-7 atmosphere it is seen that with 125 percent of the MEM mass the relevant portion of the heating-time history can be accurately repeated, and for the VM-8 the mass should be 75 percent that of the MEM.

HEATING RATE SCALING

MARS

$V_E = 11{,}050$ FPS

$\gamma_E = -7°$

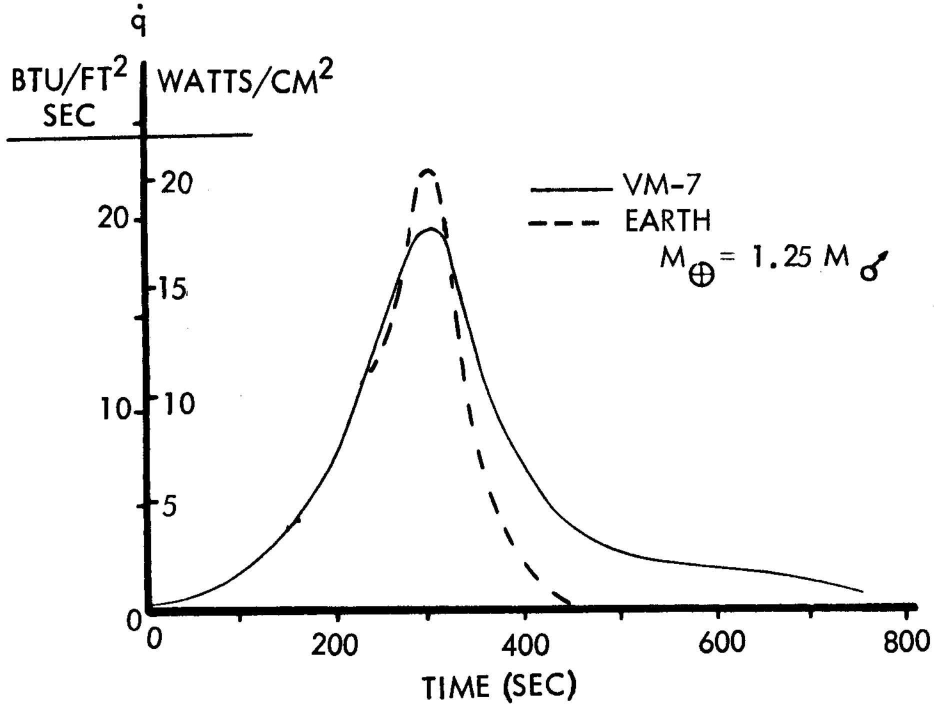
$M/C_L A = 6.4$ SLUGS/FT^2

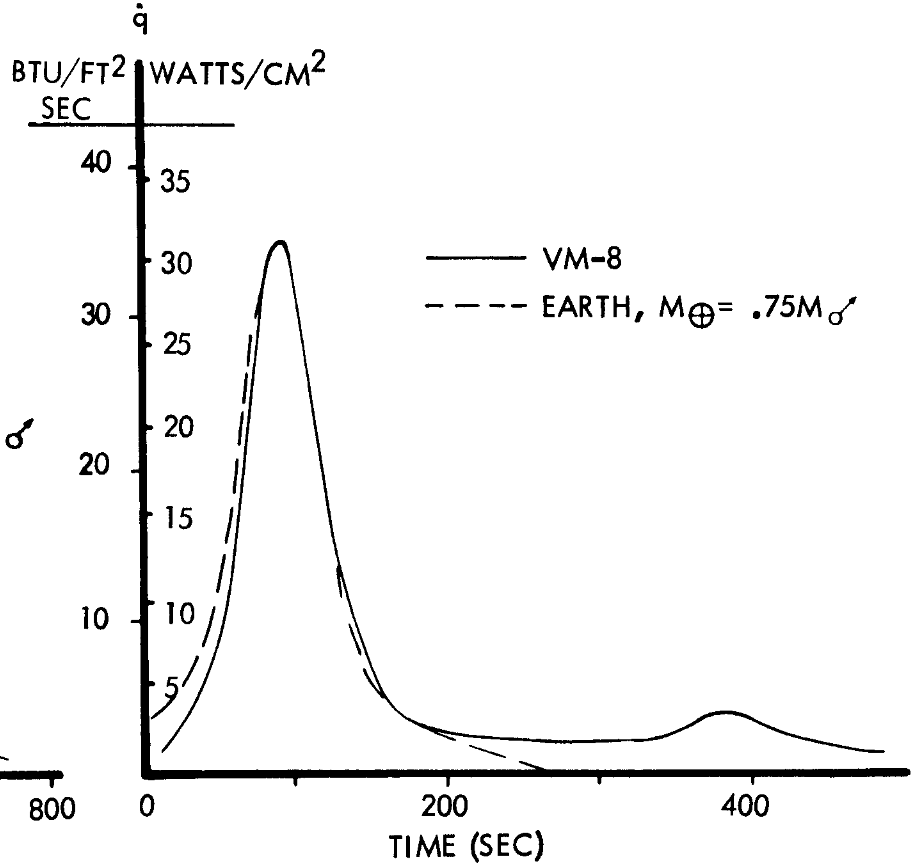
EARTH

AT PEAK HEATING

$V_{\oplus} = 1.07\, V_{♂}$

$$\left(\frac{M}{C_D A}\right)_{\oplus} \sin \gamma_{\oplus} = -3.18 \left(\frac{\rho}{\beta}\right)_{♂}$$

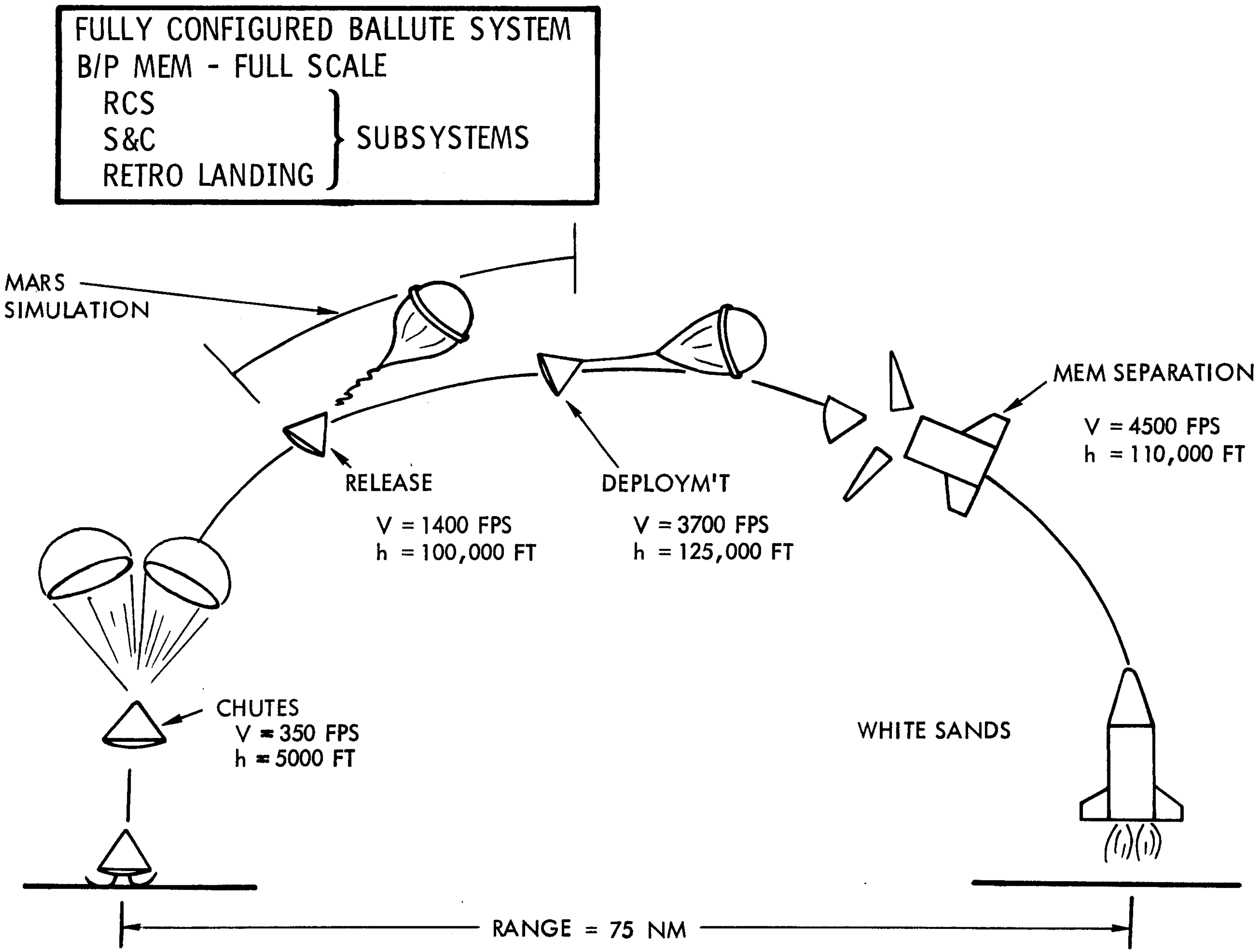




BALLUTE QUALIFICATION TESTS

The ballutes, if required, can be qualified by Earth atmospheric flights. To match Mars conditions, deployment should occur at an altitude of 125,000 ft (38 km) and a speed of 3700 fps (1.13 km/sec) on a horizontal flight path. A full-scale boilerplate MEM can be used, but it must have a stabilization and control system to orient and stabilize the vehicle before ballute deployment. A special recovery system consisting of a cargo parachute subsystem with altitude and velocity sensors and a landing gear also would be required. Launch would be from White Sands on a 33-ft (10-m) diameter solid booster of the Little Joe type. In order to achieve the required test conditions, the test vehicle would separate from the booster at an altitude of 110,000 ft (33.5 km) and a velocity of 4,500 fps (1.37 km/sec).

BALLUTE QUALIFICATION TESTS*



*IF REQ'D

SCALING LAWS

PARACHUTE/BALLUTE TESTS

Scaling laws were derived for testing of full-size or scaled parachutes and ballutes at Earth. The principle used is that all the forces encountered (i. e., inertial, steady and unsteady aerodynamic and elastic forces) should be in the same proportion to each other in the scaled Earth test as at Mars. Since the MEM ballutes would be deployed at a zero flight path angle, gravitational effects would not be significant; therefore, gravity forces were not considered. The Mach number and Reynolds number should be matched for aero-dynamic similitude. Only approximate matching of the Reynolds number is required to assure like flow patterns and aerodynamic coefficients.

The analysis resulted in the scaling laws shown. By matching Mach number, the velocities also are nearly matched (because sonic velocities at Mars are nearly the same as at Earth). The altitude at Earth is chosen to match the ambient densities. Test vehicle weight decreases proportionately to the cube of the scale; e. g., from 75,000 pounds (34,000 kg) for full scale to 9400 pounds (4300 kg) for one-half scale and 2800 pounds (1270 kg) for one-third scale.

SCALING LAWS

PARACHUTE/BALLUTE TESTS

REQUIREMENTS

MAINTAIN SAME RATIO OF FORCES IN MARS & EARTH TESTS

- INERTIAL
- STEADY AERO
- UNSTEADY AERO
- ELASTIC

SAME MACH NUMBER & REYNOLDS NUMBER

SCALING LAWS

	PROPORTIONAL TO	MARS	EARTH TESTS		
			FULL SIZE	1/2 SCALE	1/3 SCALE
DIAMETER (FT)	D	60	60	30	20
WEIGHT (LBS)	ρD^3	109,000	109,000	13,600	4,000
DENSITY (SLUGS/FT^2)	ρ	10^{-5}	10^{-5}	10^{-5}	10^{-5}
ALTITUDE (FT)	--	30,000	125,000	125,000	125,000
VELOCITY (FPS)	c	3,400	3,700	3,700	3,700
TIME (SEC)	D/c	1	0.92	0.46	0.31

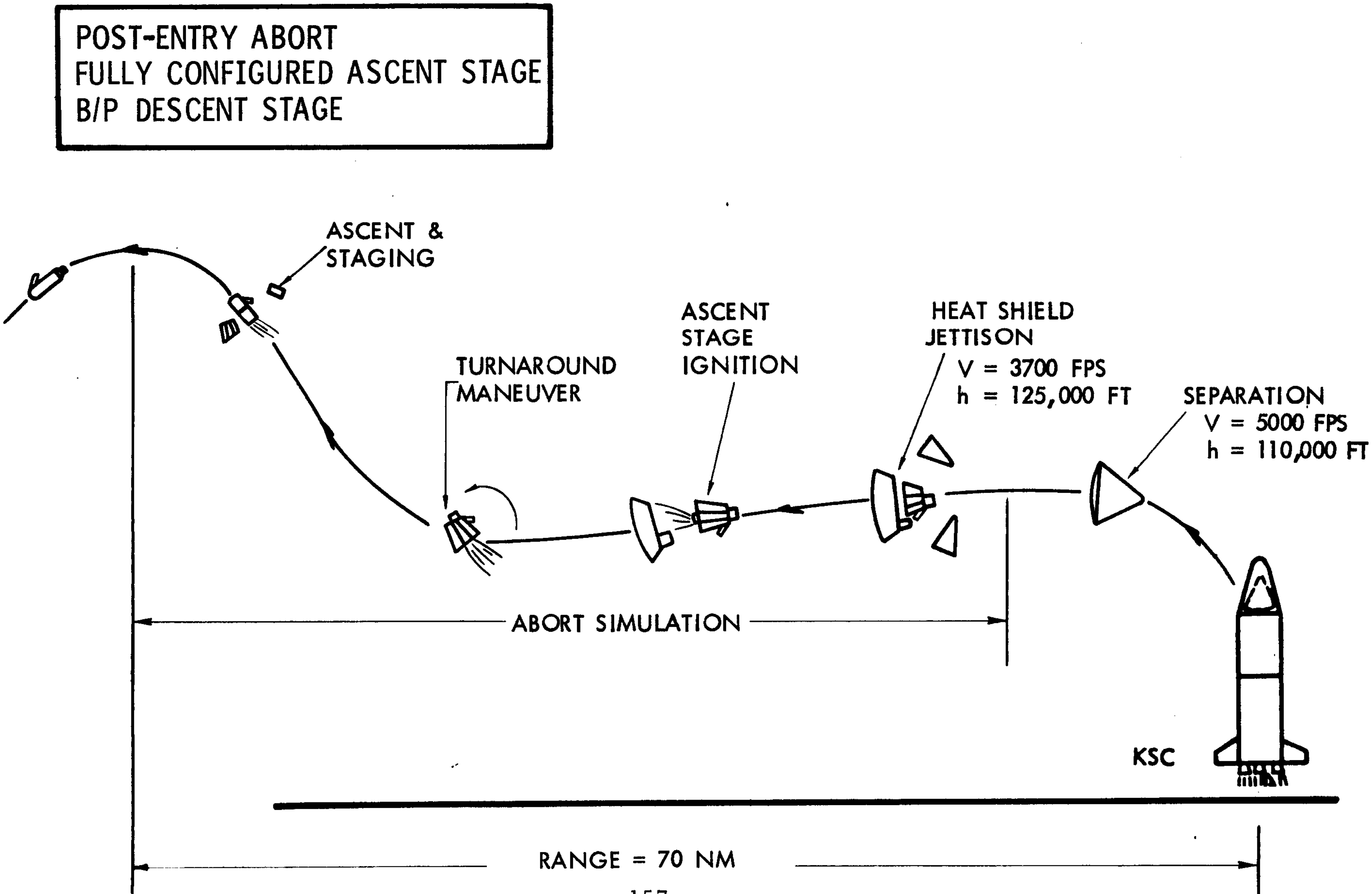
UNMANNED FLIGHT ABORT TEST

The most critical abort situation for the MEM occurs after Mars entry but before ballute deployment or retropropulsion. Aerodynamic loads are significant, and the vehicle is oriented opposite to the direction desired for ascent back to orbit.

This situation can be simulated fully by an unmanned Earth atmospheric flight test. The test vehicle will consist of a fully configured MEM ascent stage and a boilerplate descent stage to provide the correct interfaces. The vehicle could be launched at Kennedy Spacecraft Center on an uprated Saturn I to an altitude of 110, 000 feet (33. 5 km) and velocity of 5, 000 fps (1. 5 km/sec). If a suitable solid booster has been developed, (e. g., for possible full-scale ballute tests), the abort test could be conducted at White Sands. Test conditions representative of the Mars abort would be reached at apogee with the lift vector up at which time the simulated abort test would begin. The ascent stage heat shield would be jettisoned, the ascent stage separated from the descent stage, and the ascent engine ignited. The thrust would decelerate the MEM sufficiently so that a turnaround maneuver could be performed and the vehicle then accelerated along a preprogrammed flight path to a simulated Mars ascent orbit. Since orbital velocity cannot be achieved at Earth, the ascent stage would reenter. There appears no need to recover the stage, and therefore, it would be destroyed on impact.

All phases of the abort are simulated, including the velocity, density, Mach number, and aerodynamic loads. Critical issues, which can be verified in this test, include firing of the ascent engine in a low-density atmosphere, and the dynamics and aerodynamics of the separation and the turnaround maneuver.

UNMANNED FLIGHT ABORT TEST

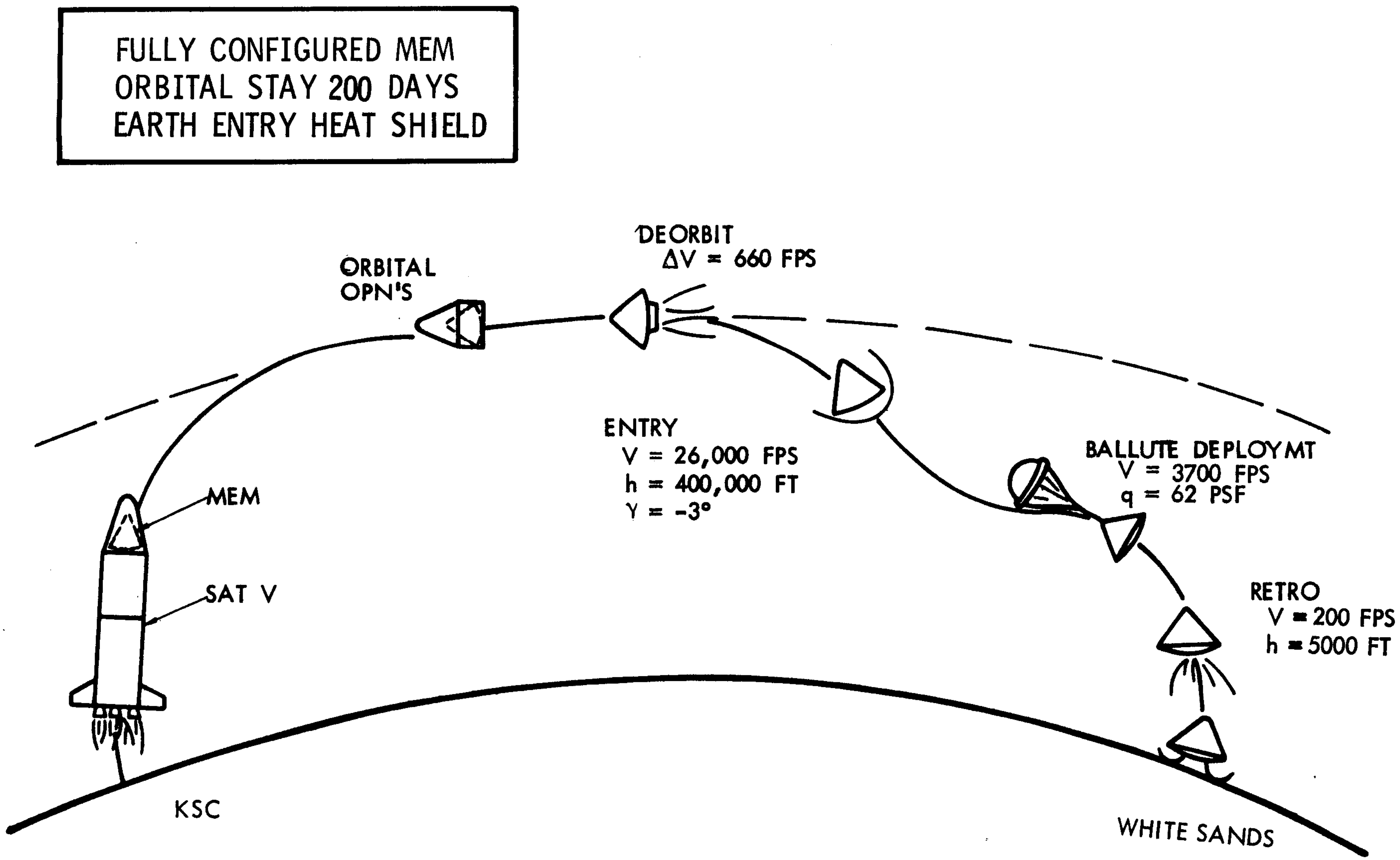


UNMANNED EARTH ORBITAL ENTRY AND LANDING TEST

A full-size, all systems-up, unmanned MEM is launched into Earth orbit by a two-stage Saturn V. The MEM is enclosed in a shroud to simulate the spacecraft and its environment. After approximately 200 days in Earth orbit, the MEM is separated from the shroud, oriented, and the deorbit motor commanded to fire. Entry occurs at 26,000 fps (7.9 km/sec) at an entry angle of -3 degrees. This test qualifies the heat shield for entry conditions which are anticipated to be considerably more severe than will be encountered during Mars entry. For the case of retropropulsive descent, the engines are ignited after terminal velocity is reached at about 5000 ft (1.5 km). For the case of ballutes (if adopted), a zoom maneuver may be performed after entry to provide representative altitude and velocity conditions for ballute deployment. The ballute may be jettisoned at high altitudes [e.g., 100,000 ft (30 km)] or retained to a few thousand feet before firing the descent motor for landing. Hover capability of one to one and a half minutes is available for either mode.

This test provides a MEM integrated subsystem test through all phases to landing. The ascent stage will be available for inspection and ground tests to evaluate the effects of the entry and landing operations on subsystem performance (e.g., the ascent propulsion motors will undergo static firing). This may be preferable to a later launching with the ascent stage as part of an integrated system because considerably more data can be obtained from such a test.

UNMANNED EARTH ORBITAL ENTRY AND LANDING TEST



SCALING ANALYSIS

EARTH LANDING

Minor modification to the radar altimeter constants make it possible to use the MEM retardation and landing system to land on Earth. The ΔV, T/W, and hover times for the MEM are shown for both ballute-retro and retro-only systems. The diagram on the left shows that the ballute can be deployed at speeds and altitudes which correspond to Mars conditions; the ballute is retained, however, to an altitude of 3200 ft (1 km) where the vehicle has decelerated to 200 fps (61 m/sec). The ballute then is released and the descent engine ignited and used at full thrust. Hover conditions are reached at 2000 ft (610 m) and the engine is throttled back for touchdown. In contradistinction to the situation on Mars, only a small proportion of the descent propellant is used to decelerate from the terminal velocity; most of the propellant is used for hover because of the higher gravity at Earth. The hover time at Earth is 1 to 1.5 minutes compared to 1 to 2 minutes at Mars.

With all retropropulsive descent, the engine is ignited at 4500 to 5000 ft (1.4 to 1.5 km) at a velocity of 300 to 375 fps (91 to 114 m/sec), depending on whether the low L/D or lifting-body MEM is being considered. Zero velocity is again reached at 2000 ft (610 m) and 1.5 to 2 minutes hover time is available.

SCALING ANALYSIS

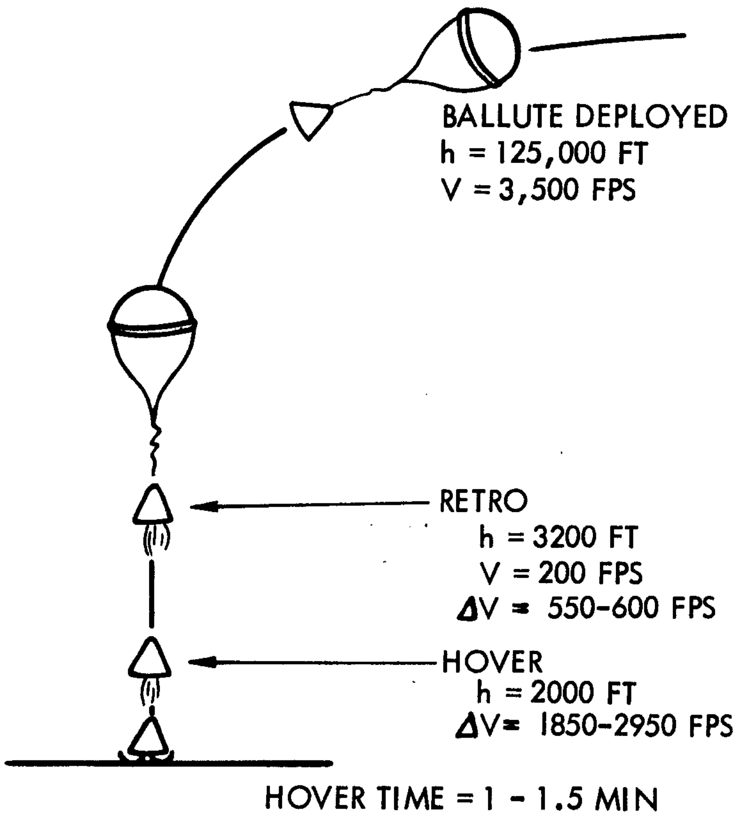
EARTH LANDING

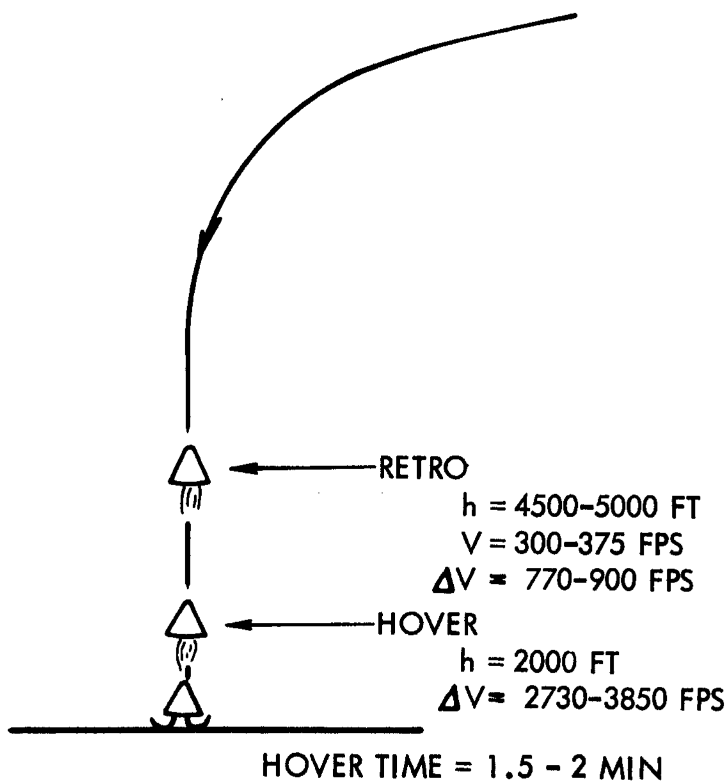
OBJECTIVES

- SIMULATE MARS RETARDATION & LANDING ON EARTH
- USE MEM SUBSYSTEMS & PROPELLANTS

MEM DESIGN CHARACTERISTICS

	BALLUTES/RETRO	RETRO
$\Delta V_{AVAILABLE}$ (FPS)	2400-3550	3500-4750
$T/W_{\oplus}$	1.5	1.5
HOVER TIME ON MARS	1-2 MIN	1-2 MIN





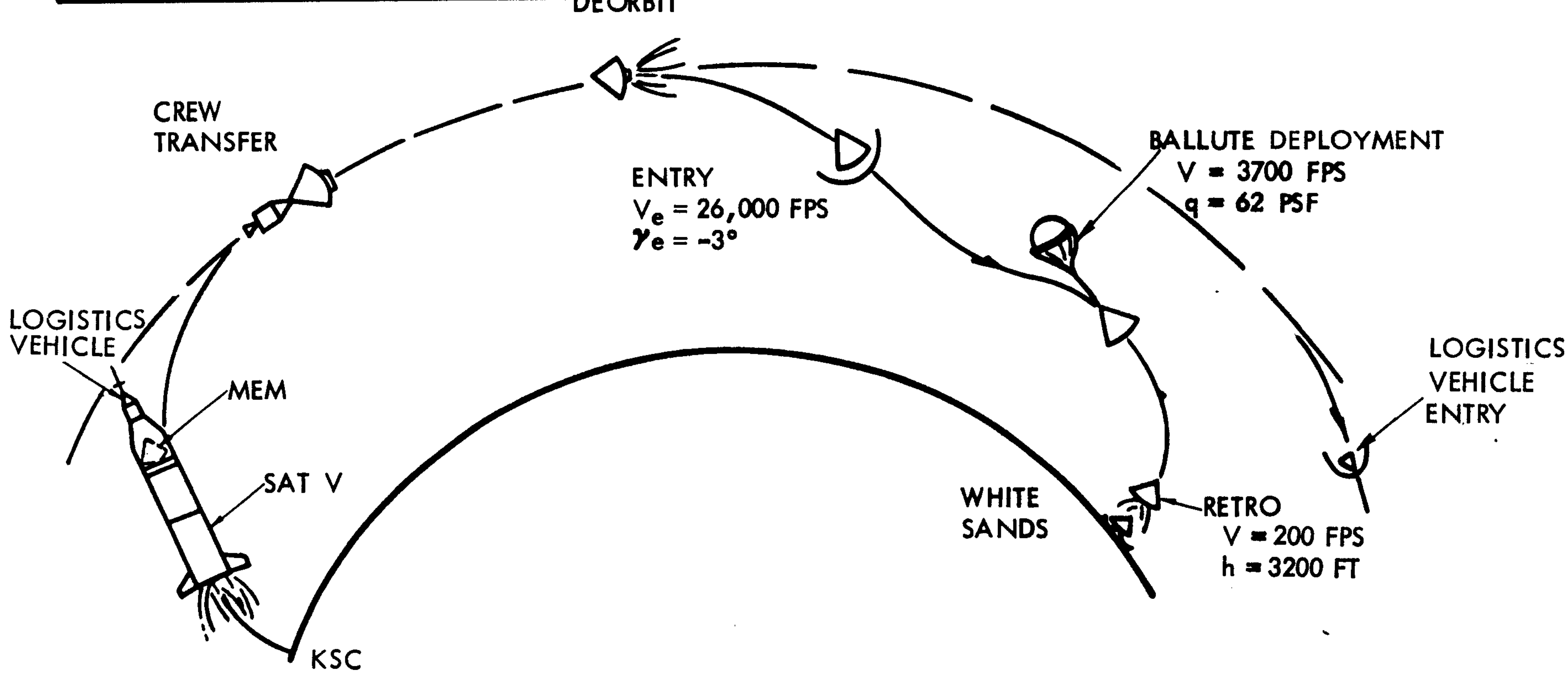
MANNED EARTH-ORBITAL ENTRY AND LANDING TEST

This test is essentially a manned version of the unmanned entry and landing test. Since subsystem operation after a long exposure to space already has been verified in the unmanned test, the duration of this test can be limited to approximately 10 days to reach equilibrium and for checkout before entry.

A three-man crew would be launched in an Apollo CSM on the same two-stage Saturn V as the MEM. Abort capability can be provided during boost and in orbit. Following a transposition docking maneuver, two members of the crew would transfer to the MEM and bring it back to Earth, and the third man would return the CM. If a four-man crew is required for the MEM entry, a five- or six-man version logistics carrier would be required.

MANNED EARTH ORBITAL ENTRY AND LANDING TEST

SINGLE SATURN V
FULLY CONFIGURED MEM
ORBITAL STAY 10 DAYS
EARTH ENTRY HEAT SHIELD



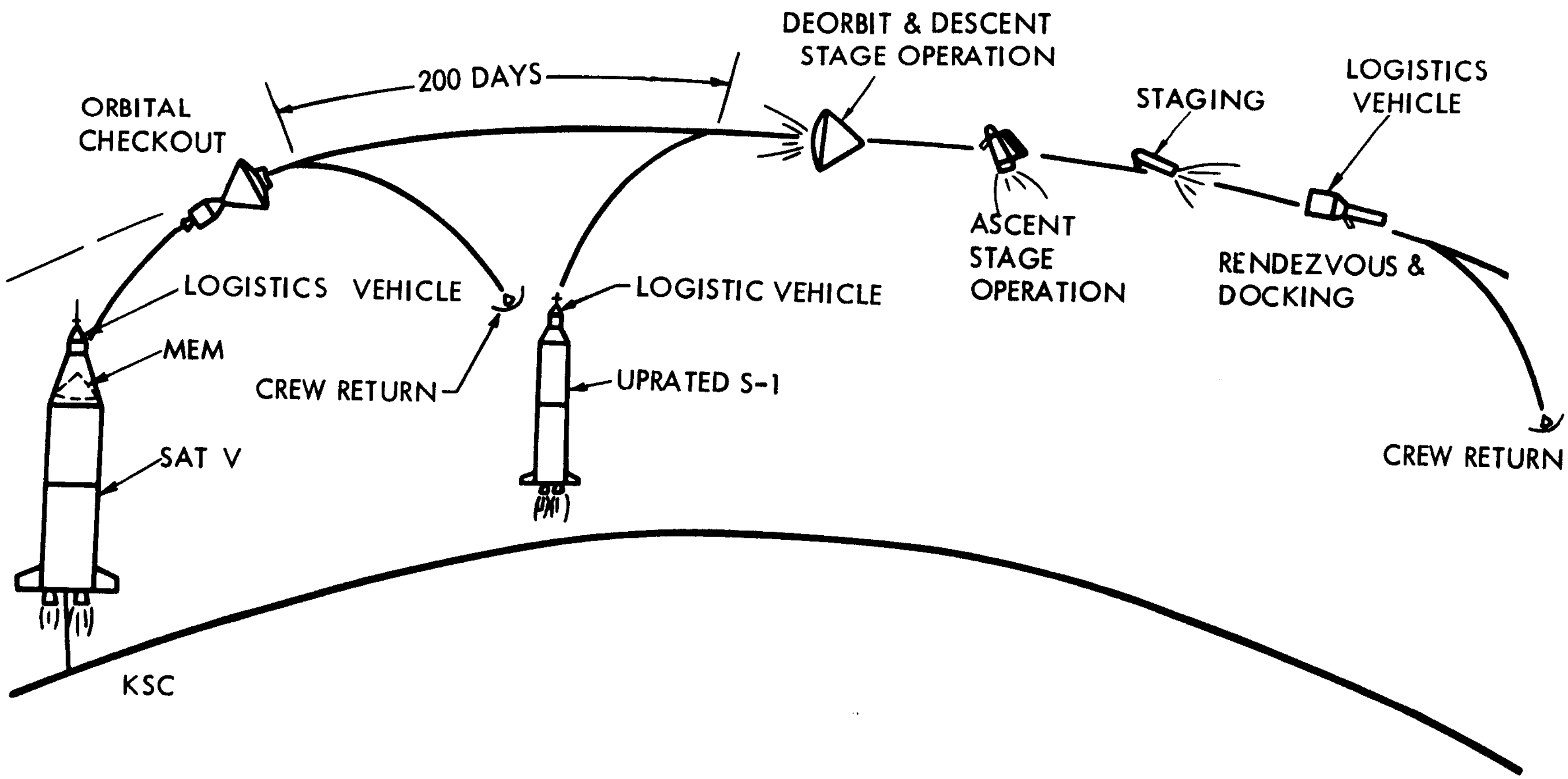
MANNED EARTH-ORBITAL PROPULSION TEST

In this test a fully configured MEM and manned CSM are launched into Earth orbit by a single Saturn V. The MEM, enclosed in a shroud which simulates the spacecraft environment, remains in Earth orbit for 200 days to simulate Mars transit conditions. At the end of the 200 days, a fresh crew mans the MEM and, after checkout, exercises the propulsion, separation, and other subsystems in the same sequence as the MEM mission. The propulsion operations are programmed so as to return the MEM to the same orbit as the CSM with a separation distance of approximately 140 NM (260 km). Rendezvous and docking are then practiced with the CSM, after which the crew returns to Earth in the CM and the MEM is abandoned in orbit.

This is essentially a manned test of the propulsion and docking subsystems after a long space exposure. Entry, landing, and Mars surface operations are not simulated. A single Saturn V and an uprated Saturn I for each CSM are required, as well as a shroud to simulate the spacecraft. No MEM hardware is recovered.

MANNED EARTH ORBITAL PROPULSION TEST

MEM MANNED DURING PROPULSION SYSTEM OPERATION
FULLY CONFIGURED MEM
ORBITAL STAY 200 DAYS
NET $\Delta V = 0$



MANNED LUNAR LANDING AND ASCENT

A manned lunar landing and ascent with the MEM has been considered as a simulated test of the Mars landing and ascent. An unmanned MEM, together with a propulsion module, is launched on a single Saturn V and is placed into orbit around the moon by the propulsion module. A manned logistics carrier later is launched to the moon, achieves the same orbit, and rendezvous with the MEM. The manned MEM propulsively descends and lands on the lunar surface, consuming both the descent and first-stage ascent propellants for this maneuver. After an appropriate stay on the surface, the MEM ascends for rendezvous with the logistics carrier, using the second-stage ascent propellants. The crew returns to Earth in the logistics vehicle and abandons the MEM in lunar orbit.

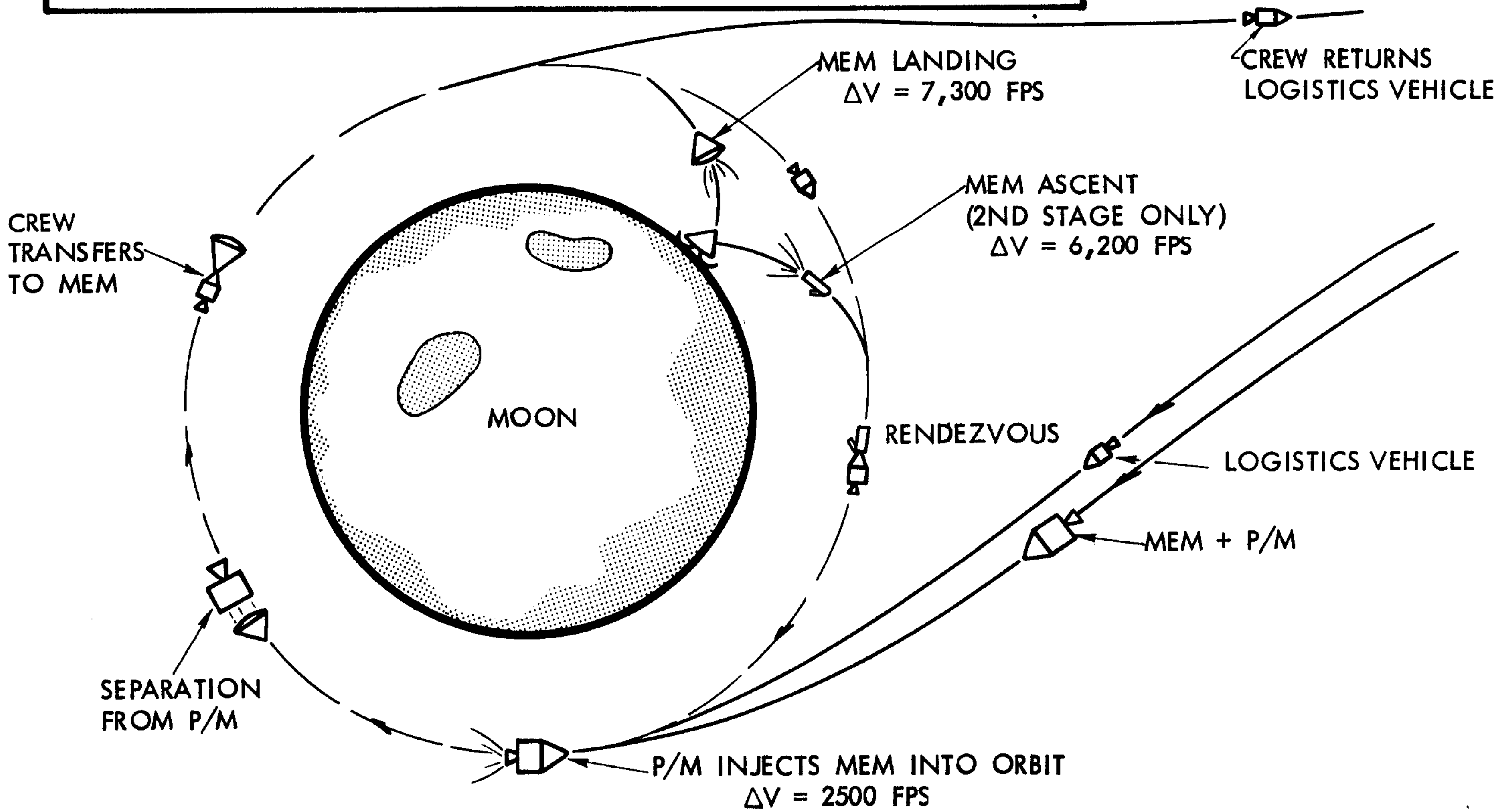
This test also could be performed as part of an integrated spacecraft system test, in which case the spacecraft takes on the role of the logistics vehicle. The lunar landing and ascent are exactly as described.

In this test the atmospheric effects during entry, surface operations, and ascent are, of course, not simulated. The test requires two Saturn V's (or more if performed with the integrated spacecraft) and a propulsion module for the MEM. MEM modifications also are required so that the first-stage ascent propellants can be used for descent.

MANNED LUNAR LANDING & ASCENT

DUAL S-V LAUNCH
- UNMANNED MEM & PROPULSION MODULE (P/M)
- LOGISTICS VEHICLE

MEM PROPULSION
- LUNAR ORBIT INSERTION - MEM & P/M
- DESCENT - MEM DESCENT & 1ST STAGE ASCENT PROPELLANT
- ASCENT - 2ND STAGE ASCENT PROPELLANT



SCALING ANALYSIS

LUNAR LANDING/ASCENT

In order to simulate a Mars landing and ascent at the moon, the ΔV available for the landing and ascent phases must match the ΔV requirements. The ΔV available to the MEM (for the nominal Mars mission assuming a low circular parking orbit) and the ΔV required for the lunar mission, also assuming a low circular orbit, are shown. It is seen that even for retropropulsive descent and maximum allowance for hover, the MEM descent stage lacks about 2,500 fps (760 m/sec) ΔV capability. The ascent stage, on the other hand, contains more propellant than is required.

In order to achieve a lunar landing and ascent it may therefore be necessary to re-apportion the descent and ascent propellants. Fortuitously, the ΔV requirements can be matched by using the MEM descent and first-stage ascent propellants for lunar descent, and the second-stage ascent propellants for lunar ascent.

SCALING ANALYSIS
LUNAR LANDING/ASCENT

OBJECTIVES

SIMULATE MARS LANDING & ASCENT ON MOON

Δ V REQUIREMENTS

LOW CIRCULAR ORBIT

(FPS)

	MARS	MOON
RETRO DESCENT & LANDING	3500 - 4750*	7300
ASCENT	16,000	6200

* INCLUDES 1-2 MINUTE HOVER

REQUIRES REDISTRIBUTION OF DESCENT/ASCENT PROPELLANTS

INTEGRATED SYSTEMS S/C CISLUNAR FLIGHT

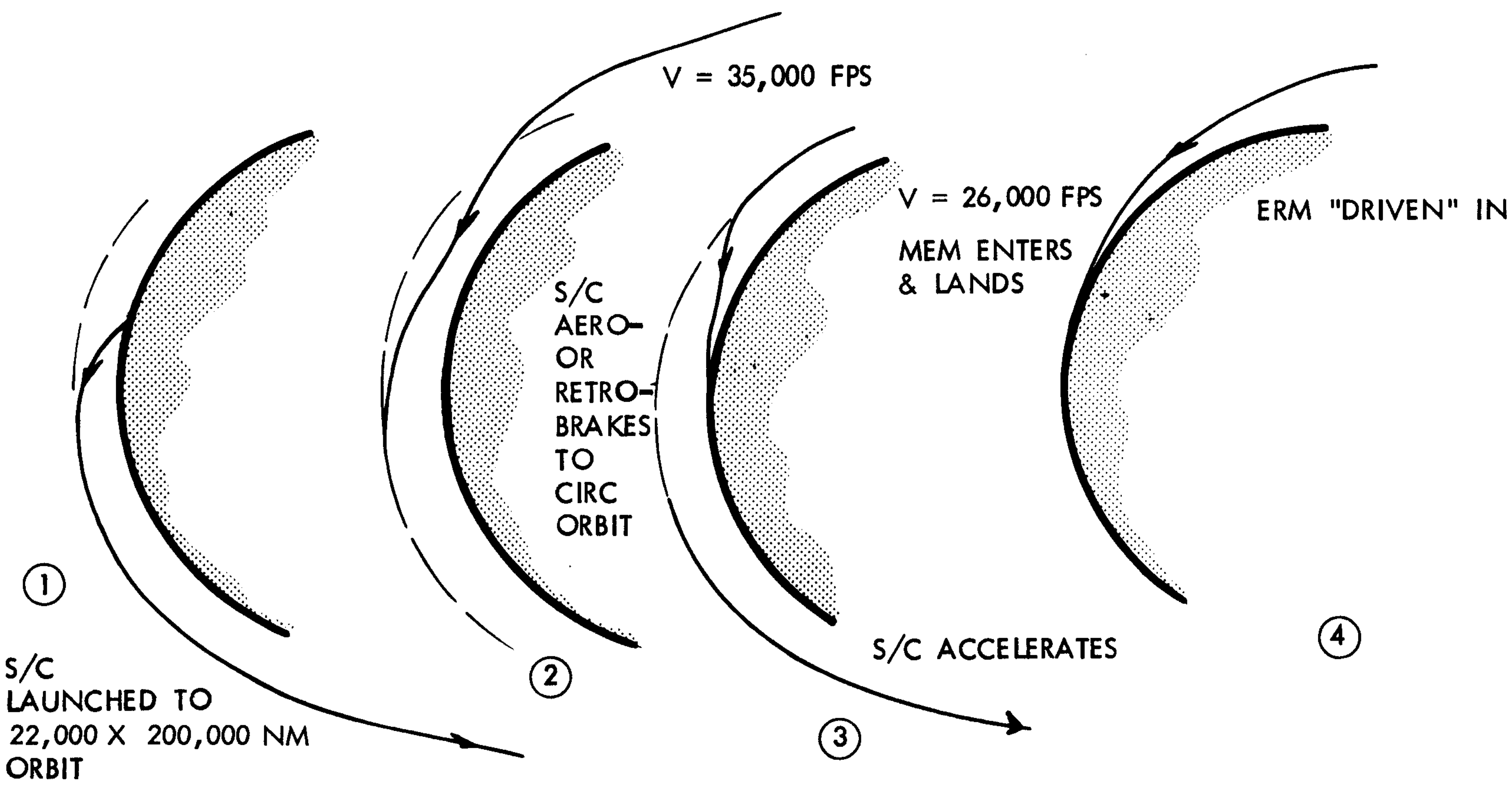
A manned integrated system spacecraft flight can be used as a dress rehearsal for the manned Mars landing mission. A fully configured spacecraft with all the component modules is assembled in Earth orbit using multiple Saturn V launches. The Earth orbit escape propulsion stage is used to inject the spacecraft into a highly elliptic cislunar orbit for the same elapsed period as the trans-Mars transit phase. At the end of this time, the spacecraft aerobrakes or retro-brakes into a low circular Earth orbit, the MEM (piloted by half the spacecraft crew) separates, enters, and lands to conclude the MEM test. The spacecraft can return to a highly elliptical orbit by firing its planet orbit departure stage and then "drive" the Earth return module (with the remainder of the crew) back into the atmosphere at hyperbolic speed. The spacecraft is abandoned in Earth orbit and may be used in the future as a space station.

This flight will simulate fully all phases of the manned Mars landing mission, with the exception of the MEM surface operations, ascent, and rendezvous. Abort capability is provided every few days, when the spacecraft approaches Earth. Rescue also is possible.

The MEM test requirements satisfied by this flight also can be satisfied as described earlier, and such tests would indeed be carried out before the MEM is committed to such a complex flight. The requirement for this flight, therefore, would be determined by the need for an integrated spacecraft system test.

INTEGRATED SYSTEMS S/C
CIS-LUNAR FLIGHT

FULLY CONFIGURED S/C
MULTIPLE S-V LAUNCHES
ALL S/C PHASES TESTED
EXCEPT MEM/S/C RENDEZVOUS
CREW DIVIDED BETWEEN MEM & ERM



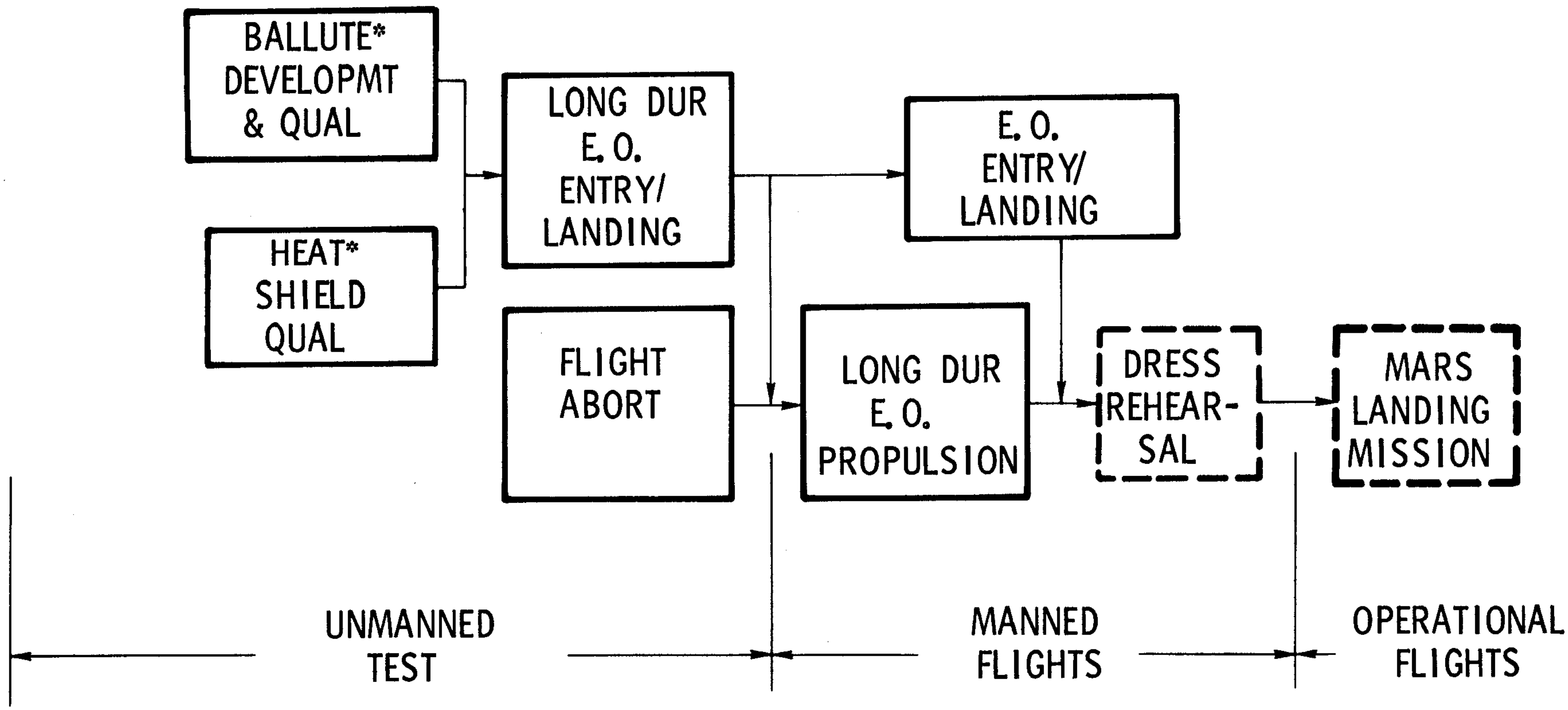
RECOMMENDED TEST PROGRAM

The recommended test program satisfies all test requirements at both the subsystems and integrated systems level with a minimum number of flight tests. Although retropropulsive descent and an Earth orbital entry heat shield design are recommended, the possibility of using ballutes and/or a Mars entry heat shield design is recognized.

The test program begins with ground testing at the component and subsystem level and builds up to ground testing at an integrated system level using "house" MEM's. The flight test program would start with development and qualification of the ballutes and qualification of the Mars heat shield, if these were incorporated in the MEM. Unmanned Earth entry and landing flight tests then would be conducted to demonstrate operation of the descent stage subsystems after extended times in Earth orbit (to simulate space soak during Mars transit) and to qualify the Earth orbital entry heat shield. This test would be followed by a manned flight to obtain entry and descent experience; extended stay times in orbit are not required. The other branch of the flight test program would start with an unmanned simulated abort test to verify the dynamics and aerodynamics of pre-landing abort as well as ascent engine operation. The next Earth orbital flight would be manned and test the descent and ascent propulsion subsystems, exercise of the rendezvous and docking maneuver, and the long-duration capability of the MEM would be demonstrated.

After completion of these tests, the MEM would be fully developed and qualified. A possible "dress rehersal" flight of the integrated spacecraft, including the MEM in Earth orbit, might be considered before committing the Mars landing mission.

RECOMMENDED TEST PROGRAM



*IF BALLUTES & MARS ENTRY HEAT SHIELD ARE USED

TEST REQUIREMENTS SATISFIED BY RECOMMENDED TEST PROGRAM

Test requirements for a subsystem are considered satisfied if the subsystem has been either (a) flight-tested under simulated MEM mission environments in its dormant and active phases, or (b) ground-tested in an active or dormant phase in a simulated environment. The recommended test program satisfies all of the test requirements.

TEST REQUIREMENTS SATISFIED BY RECOMMENDED TEST PROGRAM

OPERATIONAL PHASE

SUBSYSTEM	ACCEPTANCE	PRELAUNCH	BOOST	E.O. OPERATIONS	MARS INJECTION	TRANSIT	MARS CAPTURE	ORBITAL CHECKOUT	SEPARATION	DEORBIT	ENTRY	TERMINAL DESCENT	LANDING	MARS SURF. OPERATIONS	PRELAUNCH CM CHECKOUT	ASCENT	RENDEZVOUS	DOCKING
RCS - DESCENT	O	- - -	- - -	- - -	- - -	- - -	- - -	- - -	- - -	- - -	●							
ORBITAL	O	- - -	- - -	- - -	- - -	- - -	- - -	- - -	●	O	- - -	- - -	- - -	- - -	- - -	●	●	O
PROPULSION - RETRO	- - -	- - -	- - -	- - -	- - -	- - -	- - -	- - -	- - -	●								
- DESCENT	O	- - -	- - -	- - -	- - -	- - -	- - -	- - -	- - -	- - -	- - -	●						
- ASCENT	O	- - -	- - -	- - -	- - -	- - -	- - -	- - -	- - -	- - -	- - -	- - -	- - -	- - -	- - -	●		
GUIDANCE AND CONTROL	O	- - -	- - -	- - -	- - -	- - -	- - -	O	O	O	●	O	●	- - -	O	●	●	O
THERMAL PROTECT - H/S	- - -	- - -	- - -	- - -	- - -	- - -	- - -	- - -	- - -	- - -	●							
- INSUL	- - -	O	O	O	O	●	O	O	O	O	●	O	O	●	O	O	O	O
POWER SYSTEM	O	- - -	- - -	- - -	- - -	- - -	- - -	O	O	O	O	O	O	●	O	O	O	O
ECLSS	O	- - -	- - -	- - -	- - -	- - -	- - -	O	O	O	O	O	O	●	O	O	O	O
COMMUNICATIONS	O	- - -	- - -	- - -	- - -	- - -	- - -	O	O	O	●	O	O	O	O	O	O	O
BALLUTES/CHUTES	- - -	- - -	- - -	- - -	- - -	- - -	- - -	- - -	- - -	- - -	- - -	●						
MECHANISMS - DOCKING	- - -	- - -	- - -	- - -	- - -	- - -	- - -	- - -	- - -	- - -	- - -	- - -	- - -	- - -	- - -	- - -	- - -	O
- LANDING GEAR	- - -	- - -	- - -	- - -	- - -	- - -	- - -	- - -	- - -	- - -	- - -	- - -	●					
- SEP DEVICE	- - -	- - -	- - -	- - -	- - -	- - -	- - -	- - -	O	●	- - -	●	●	- - -	- - -	●		
STRUCTURE	- - -	- - -	●	- - -	O	- - -	●	- - -	- - -	- - -	●	●	●	- - -	- - -	●	- - -	O
AERODYNAMIC CONFIG											●	●						
CHECKOUT SENSORS	O	- - -	- - -	O	- - -	- - -	- - -	O	- - -	- - -	- - -	- - -	- - -	- - -	O			
TYPICAL TIMELINE (DAYS)	-90	-30	0	40	40	240	240	241	242	242	242	242	242	271	272	272	272	272

- - - DORMANT
O ACTIVE
● MAJOR DEV'T ISSUES

RECOMMENDED TEST PROGRAM EVALUATION

The recommended test program simulates all mission phases and can demonstrate system operation after prolonged space exposure. The crew will be provided with pertinent experience in all phases of the mission.

This program is capable of verifying the adequacy of the MEM design to perform to the design criteria. Although the baseline Earth-orbital flight test program does not demonstrate entry, landing, and ascent in the actual Mars environment, no such requirement has been identified because it is believed that adequate simulation can be achieved.

RECOMMENDED TEST PROGRAM EVALUATION

DOES	SIMULATE ALL MISSION PHASES
	DEMONSTRATE SYSTEMS OPERATION AFTER PROLONGED SPACE EXPOSURE
	PROVIDE PERTINENT CREW EXPERIENCE
DOES NOT	DEMONSTRATE ACTUAL MARS ENTRY, LANDING AND ASCENT

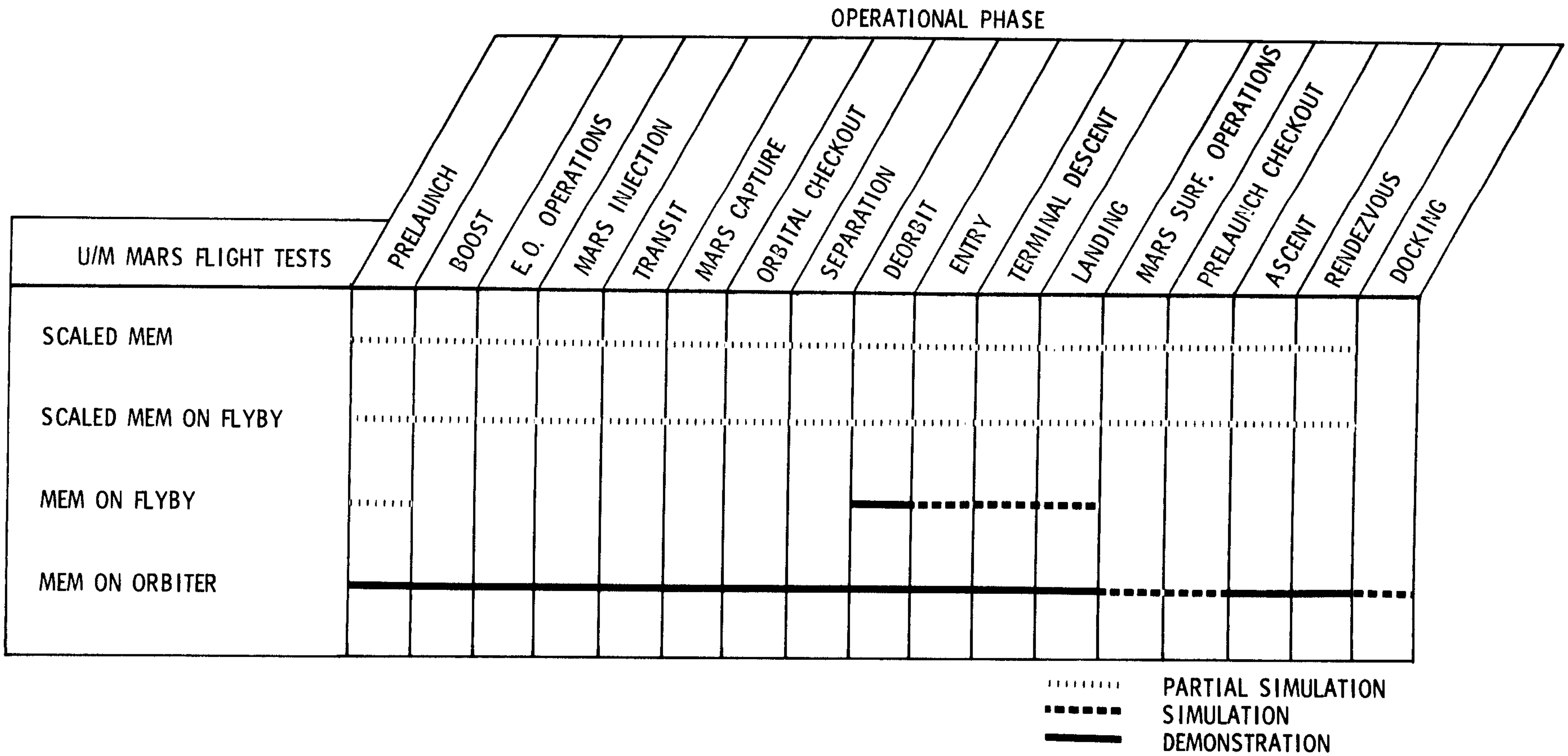
CANDIDATE UNMANNED MARS FLIGHT TESTS

SUMMARY

An analysis was made of possible unmanned flights to Mars using either subscale or full-size MEM test vehicles. Four candidate modes were examined: a scaled MEM on a single Saturn V launch, a scaled MEM on a manned Mars flyby mission, a full-size (not mass) MEM on a manned Mars flyby, and a full-scale MEM on a manned Mars orbiter. Since the scaled MEM test vehicles are in almost all respects different from the MEM vehicle, the degree of simulation that can be obtained on such tests is limited. Although the full-scale MEM vehicles also will require a relatively complex mechanical man/programmer, it is anticipated that more comprehensive simulation can be provided during the deorbit, entry, terminal descent, and landing phases. A full-scale MEM landed from an orbiter might provide the most effective flight test; however, the schedule and cost impact of such a demonstration on the ultimate manned Mars landing program must be evaluated.

It is possible to accommodate a full-size, offloaded MEM on the flyby which cannot carry enough propellant for ascent and offer only a partial demonstration. On the other hand, a MEM from an orbiter can fully demonstrate all of the operations of the manned mission except those for which a crew is essential (e. g., surface operations, checkout on Mars, and docking); these operations may be adequately simulated by an automated programmer. The term demonstration as employed here implies replication of the manned MEM mission in all relevant aspects, including exposure to the actual mission environment as opposed to simulated environments (e. g., Earth's atmosphere).

CANDIDATE U/M MARS FLIGHTS SUMMARY

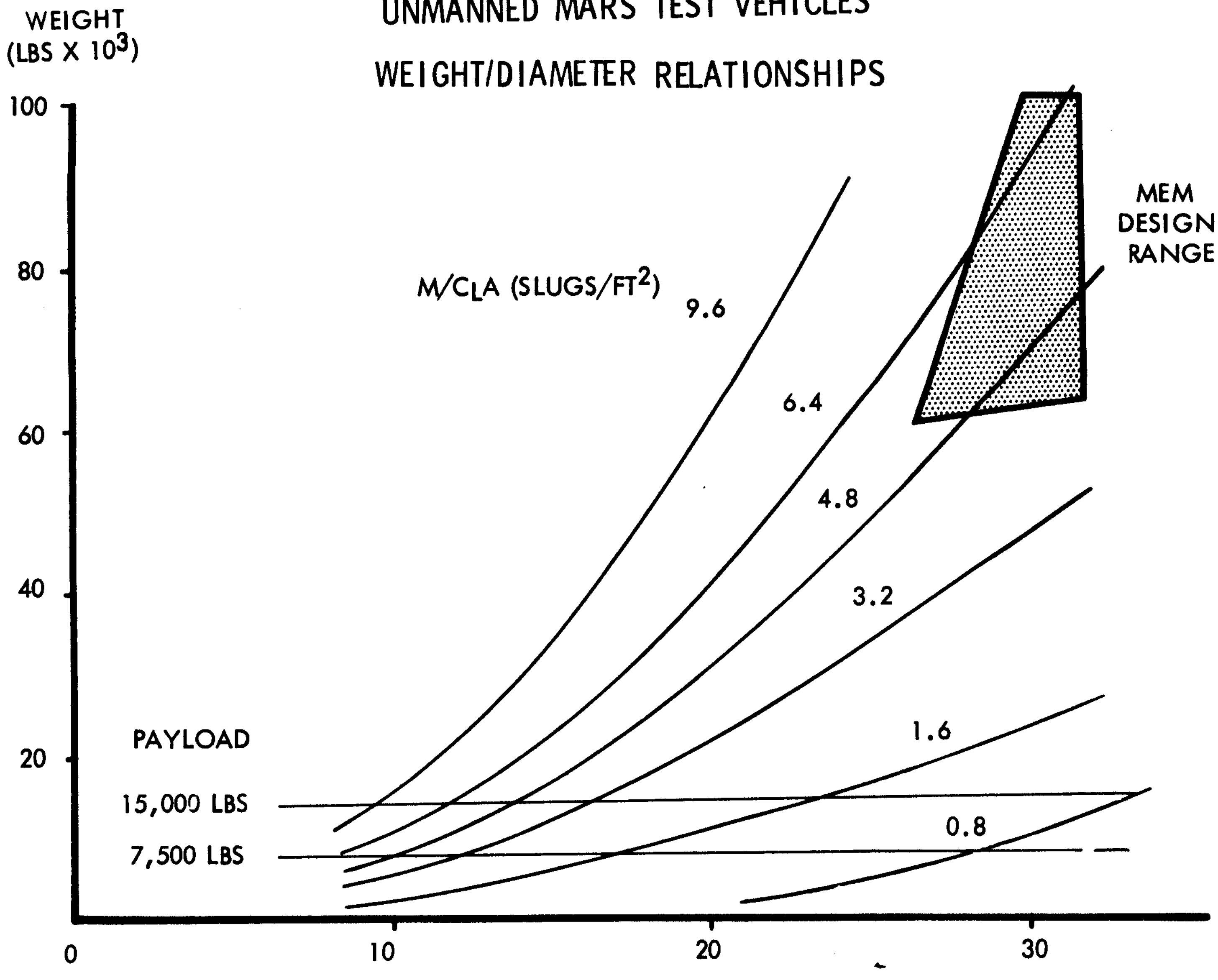


UNMANNED MARS TEST VEHICLES

WEIGHT/DIAMETER RELATIONSHIP

The relationship among weight, diameter, and M/C_LA is presented for possible MEM test vehicles. The MEM M/C_LA will fall into the range of 3.2 to 7.4 slugs/ft^2 (500 to 1160 km/m^2). To match these values, MEM test vehicles which weigh 7500 to 15,000 pounds (3400 to 6800 kg) will require diameters of 10 to 16 ft (3 to 5 m)—that is, 1/3 to 1/2 scale. Alternatively, if a full-size test vehicle of approximately 30 ft (9.1 m) is required, an M/C_LA value of only 0.7 to 1.0 slugs/ft^2 (110 to 160 kg/m^2) is realized and only limited aerothermodynamic simulation can be achieved.

UNMANNED MARS TEST VEHICLES
WEIGHT/DIAMETER RELATIONSHIPS



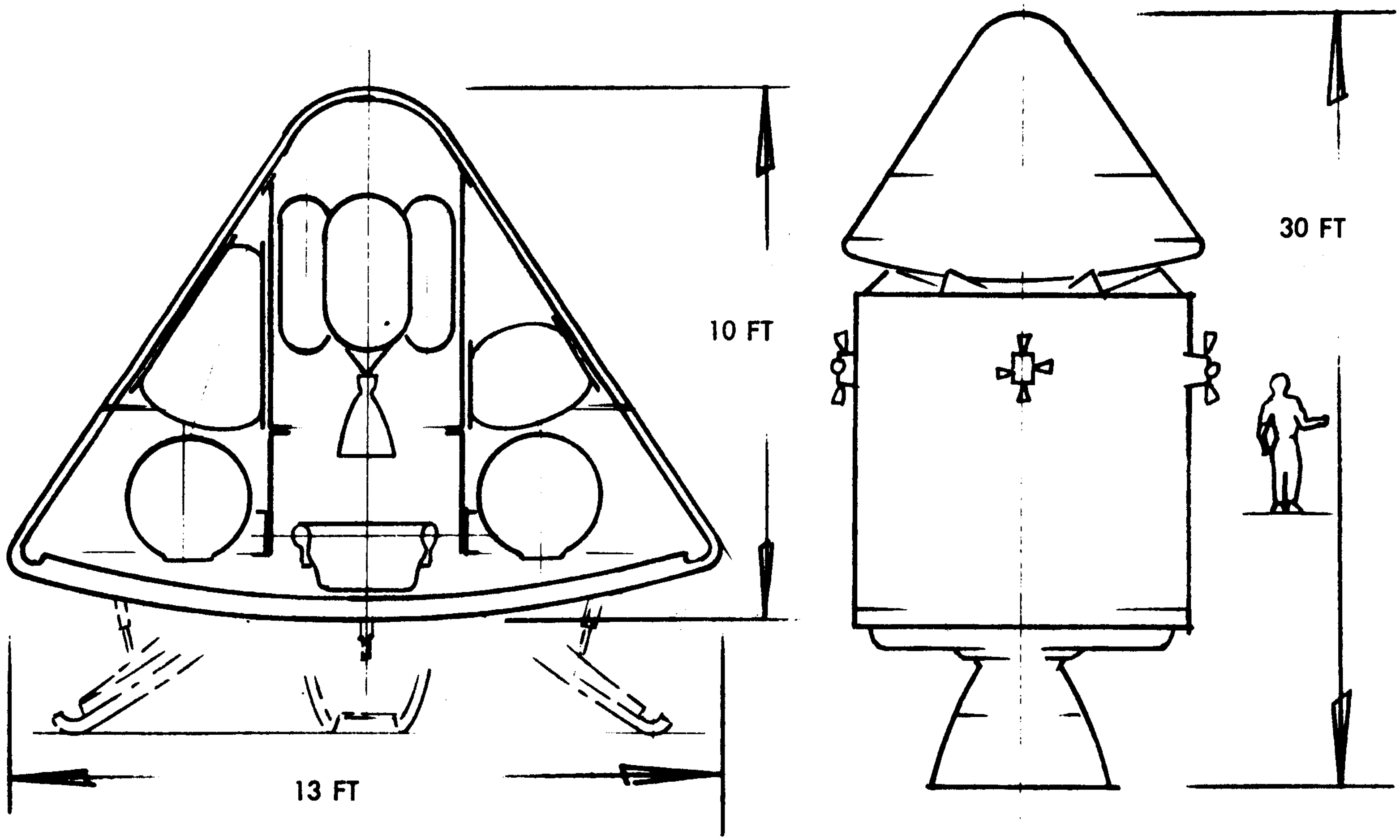
ONE-HALF SIZE TEST VEHICLE

UNMANNED MARS FLIGHT

This sketch presents a concept for an unmanned one-half size low L/D-type MEM Mars test vehicle. The dimensions and weights are similar to an Apollo command module. The structure and heat shield are of the same type as for the MEM; however, in order to use reasonably sized tanks, the internal arrangements must be considerably different. The MEM thrust ascent engine (T = 30, 000 pounds) can be used for descent and a lower thrust engine (T = 5000 pounds) used for ascent.

The Apollo SM could serve as the propulsion module for retro-braking into Mars orbit. It has approximately the right characteristics for both single Saturn V missions and for the more favorable flyby opportunities.

ONE-HALF-SIZE TEST VEHICLES U/M MARS FLIGHT



MEM TEST PROGRAM SCHEDULE

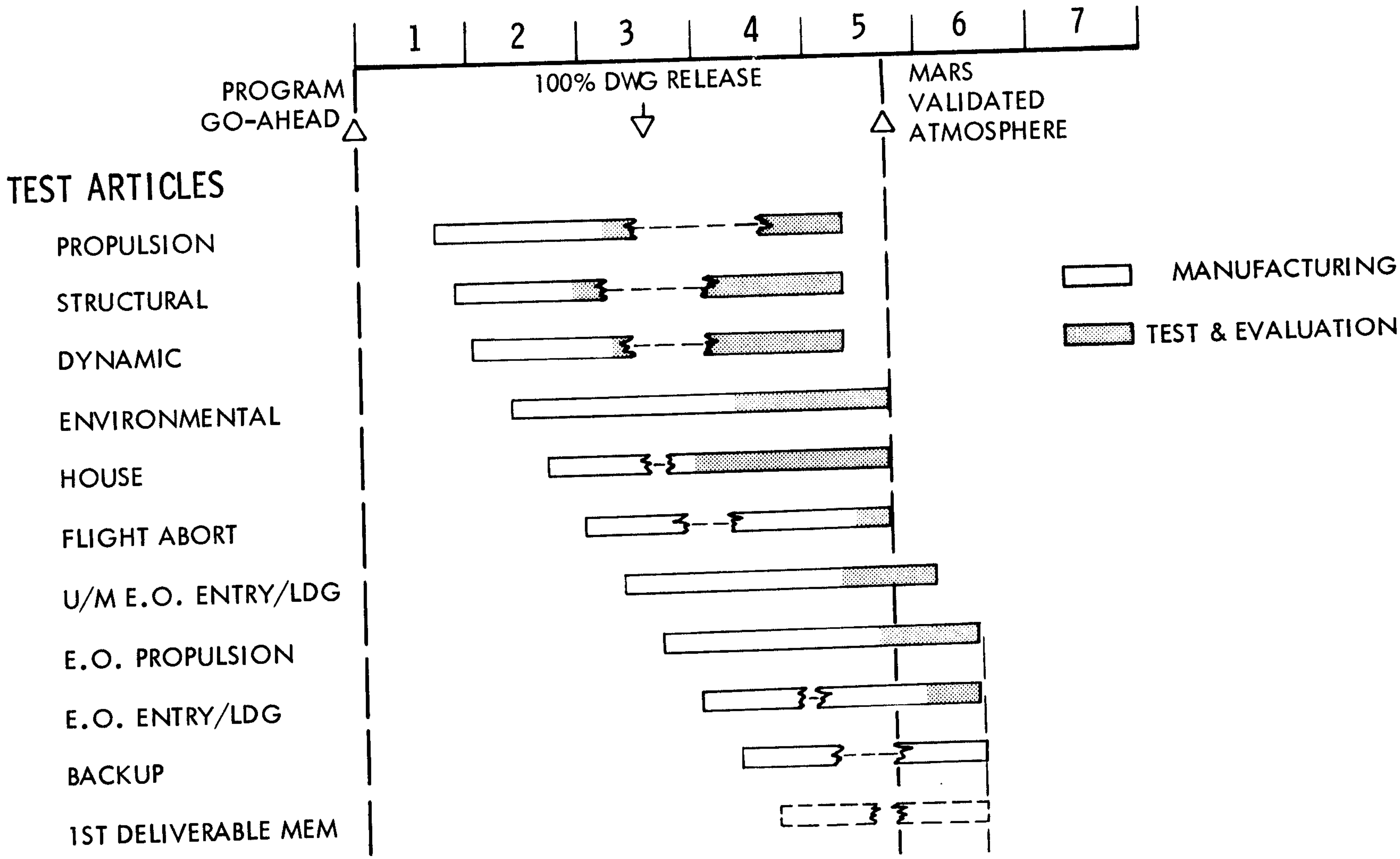
(No Ballutes/E.O. Entry Heat Shield)

The schedule for the recommended MEM development and test program is presented. Assumptions included a maximum production rate of six vehicles per year, starting with the simplest and progressing to the most complex vehicles, and a manufacturing and systems installation schedule similar to the Apollo CSM. The program is success-oriented (i.e., it is assumed that all major tests will be successful) and does not include ballute or Mars heat shield tests.

The schedule indicates that five and one-half years are required from Phase D program go-ahead to delivery of the first qualified MEM to the spacecraft integration contractor. An important constraint is that a validated Mars atmospheric model must be obtained (e.g., by a Mars surface probe) within four and three quarter years from go-ahead to prevent a delay in the program.

MEM TEST PROGRAM SCHEDULE

NO BALLUTES, E.O. ENTRY HEAT SHIELD



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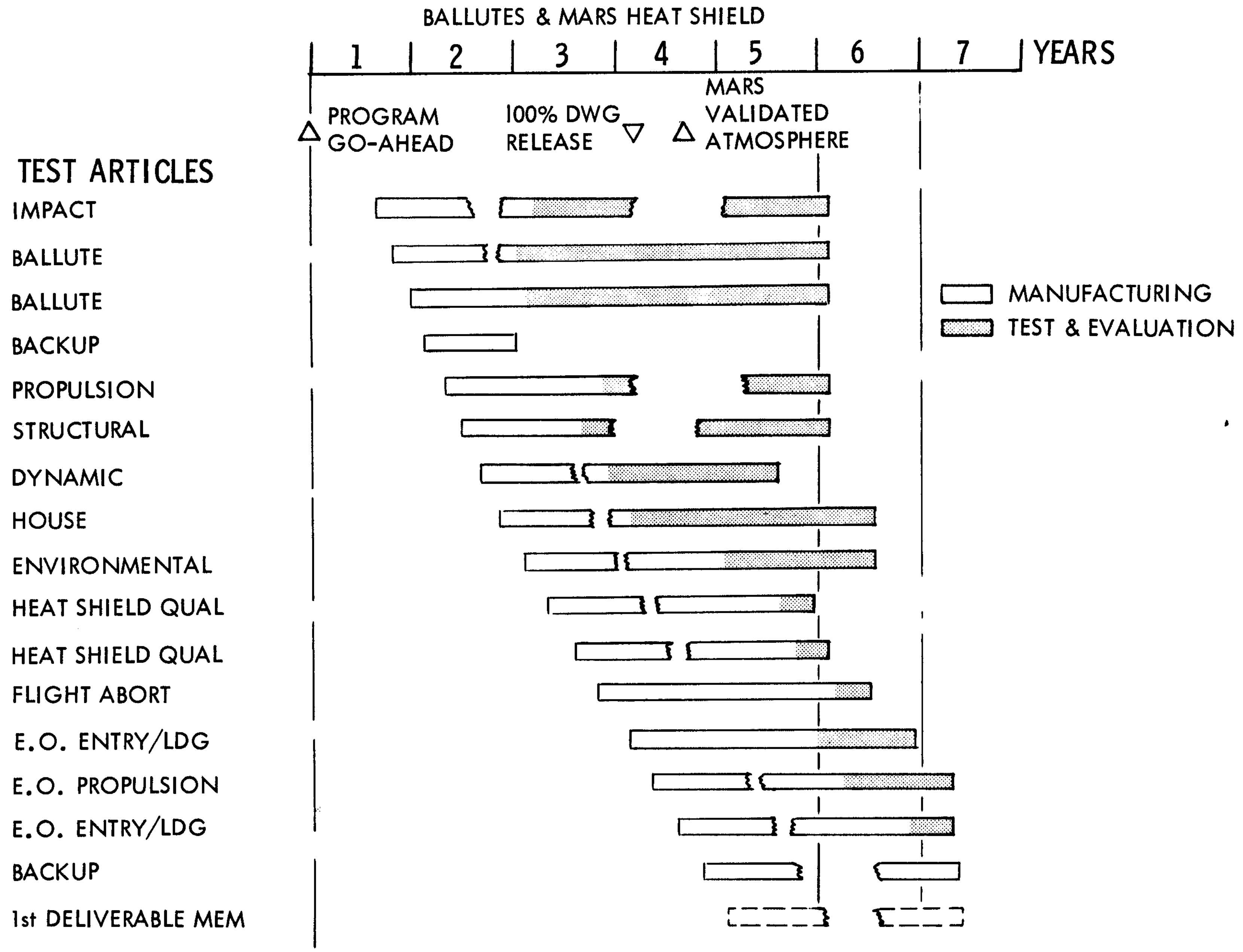
MEM TEST PROGRAM SCHEDULE

Ballutes and Mars Heat Shield

The schedule extension resulting from the use of ballutes and a Mars entry heat shield is shown. Three additional boilerplates for ballute testing, an impact test article, and two partially configured MEM's for the heat shield test will be added to the program.

The program would require 6-1/4 years from Phase D go-ahead to delivery of the first qualified MEM to the spacecraft integration contractor. Validation of the Mars atmosphere would be required 3-1/2 years from go-ahead to permit heat shield design verification and testing.

MEM TEST PROGRAM SCHEDULE

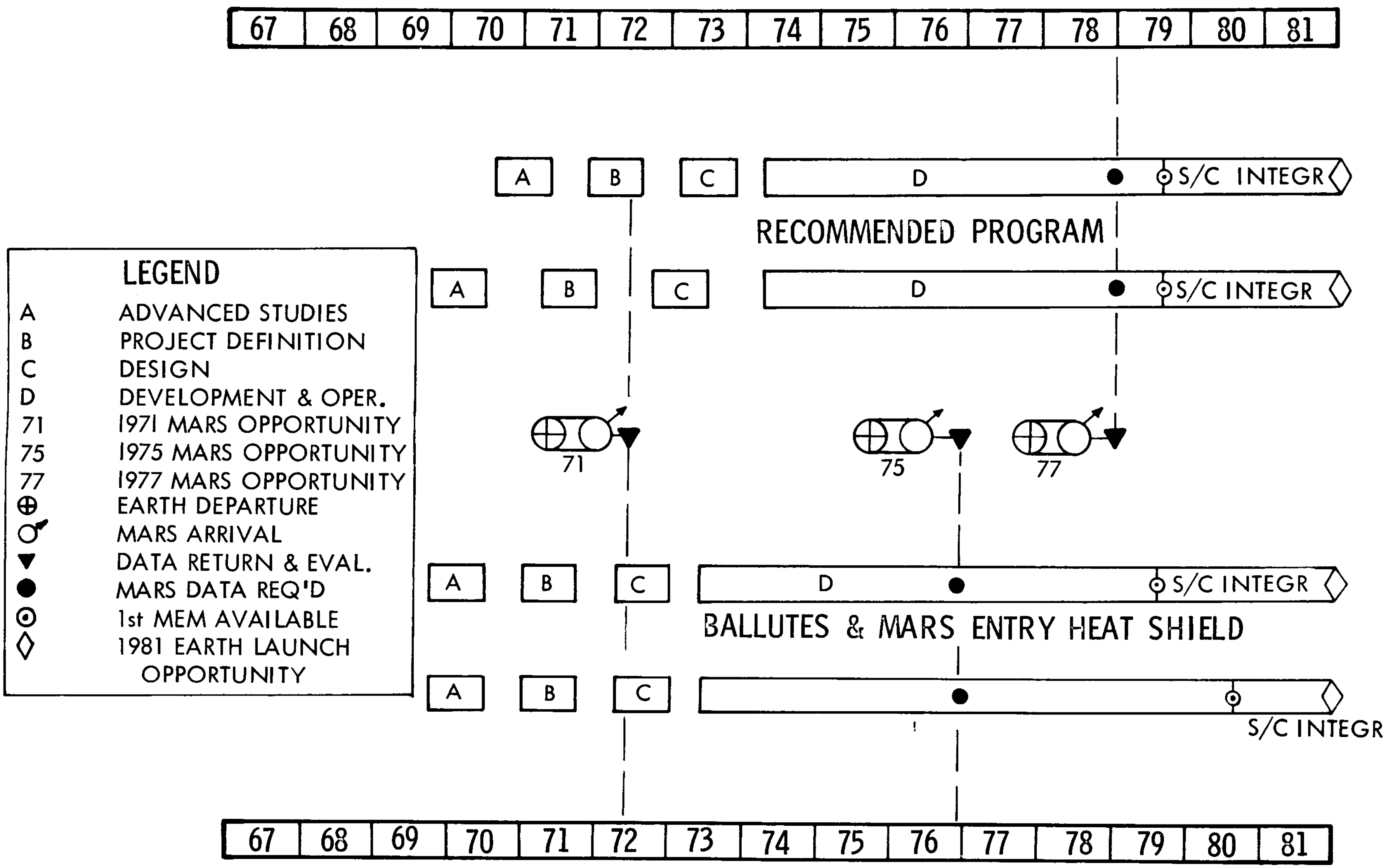


SCHEDULE PHASING FOR 1982 MANNED MARS LANDING

Three representative schedules are shown to meet the earliest postulated manned Mars landing date of 1982. For the recommended MEM design, which employs a retropropulsive descent and an Earth atmosphere entry heat shield design, the Phase A studies could begin as late as 1970, with a Phase D go-ahead in 1974, to accomplish a 1982 landing. A 2-1/4 year period has been assumed for spacecraft integration testing. If ballutes and a Mars atmosphere heat shield design are incorporated, Phase A studies leading to a Phase D program go-ahead in 1973 should be started in 1969. A later start date, or unexpected schedule delays, will defer a 1982 Mars landing.

Mars atmosphere must be validated by 1976 for the ballute/Mars heat shield program and by 1978 for the all retropropulsive retardation/Earth entry heat shield program.

SCHEDULE PHASING
FOR 1982 MANNED MARS LANDING



FACILITY REQUIREMENTS

Facility requirements for the recommended test program (retropropulsive descent/Earth entry heat shield) include the following:

a. A drop test facility capable of controlled impact tests on simulated Mars terrain, including both horizontal and vertical velocity components.

b. An environmental chamber capable of accommodating a full sized MEM. The chambers at NASA/MSC and the Arnold Engineering Development Center (AEDC) both have adequate inside dimensions; however, the doors are too small.

c. Propulsion facilities at White Sands and Las Cruces must be modified for space storable propellants (e.g., FLOX/CH_4) and vehicles as large as the MEM.

d. The Saturn V launch complex must be modified for operation of two-stage Saturn V's (S-IC and S-II) with the MEM and Apollo CSM (or logistics support vehicle) payloads. These modifications may be accomplished prior to the MEM test program for precursor programs, such as the Mars flyby.

Additional modifications which will be required at White Sands if ballutes are adopted, include facilities for the operation of large solid boosters for the ballute qualification tests.

If a Mars heat shield design is selected, the launch facilities at Kennedy Spacecraft Center must be modified to operate (1) the S-IC stage of the Saturn V as a single stage launch vehicle; or (2) a hammer head configuration of the uprated Saturn I; or (3) a launch vehicle of equal capability.

FACILITY REQUIREMENTS

EARTH ENTRY HEAT SHIELD--NO BALLUTES

FACILITY	REQUIREMENTS
DROP TEST FACILITY ENVIRONMENTAL CHAMBER PROPULSION FACILITY	50 TO 75 TON VEHICLES 33 FT (10 M) DIAMETER VEHICLES MODIFY WHITE SANDS & LAS CRUCES FACILITIES FOR SPACE STORABLE PROPELLANTS AND 33 FT (10 M) DIAMETER VEHICLES

ADD BALLUTES

MODIFY WHITE SANDS FACILITIES TO OPERATE 33 FT (10 M) DIAMETER VEHICLES WITH SOLID BOOSTERS

USE MARS HEAT SHIELD

MODIFY KSC FACILITIES FOR 1st STAGE (S-IC) SATURN V OR HAMMER-HEAD UPRATED SATURN I

COST METHODOLOGY

The MEM program cost methodology is based on the following elements:

a. Apollo experience provided the basis for estimating MEM hardware development costs. A cost of $12,650 per pound (dry weight) was derived from the Apollo development program. The MEM development costs were derived by multiplying this dry-weight cost by an equivalent number of MEM vehicles (approximately weighted for complexity) and the dry weight. Engines and ballutes were considered to be major development items to be costed separately; 1967 dollars were employed throughout.

b. Engine development costs were estimated from vendor-furnished data. Ballute development costs were based on Apollo parachute costs adjusted for the increased complexity introduced by larger payloads and deployment Mach numbers. Although the resulting ballute costs include a large uncertainty factor, they are more realistic than the cost data made available by candidate vendors.

c. Launch vehicle costs were based on NASA data for the uprated Saturn I and Saturn V vehicles. The data include hardware and launch operations costs; the latter were prorated for one- and two-stage Saturn V's in proportion to the stage hardware costs. Costs of solid boosters (for ballute and/or suborbital heatshield tests) were extrapolated from Little Joe II cost data.

d. Program support and management costs, including both NASA and integration functions, were assumed to be 10 percent of the total program costs based on the experience derived from other programs.

COST METHODOLOGY

MEM HARDWARE DEVELOPMENT

BASED ON APOLLO COSTS

COSTS = \$/LB X DRY WEIGHT X NO. OF EQUIV. VEHICLES

ENGINE & BALLUTE DEVELOPMENT EXTRA

SUBSYSTEMS

ENGINES - BASED ON VENDOR DATA

BALLUTES - BASED ON COMPLEXITY FACTOR RELATIVE TO APOLLO PARACHUTES

LAUNCH VEHICLES

UPRATED S-1 & S-V - BASED ON NASA DATA

SOLID BOOSTER - BASED ON LITTLE JOE II DATA

PROGRAM SUPPORT AND MANAGEMENT

ASSUMED 10% OF TOTAL COSTS

DEVELOPMENT PROGRAM COST SUMMARY

Typical cost breakdowns are shown for the recommended configuration. The subsystem apportionments are based on Apollo experience and related spacecraft studies. Engine development costs are incorporated in those of the propulsion subsystem.

A saving of just over $1 billion is realized if an all-retropropulsive retardation system and an Earth atmosphere entry heat shield are employed. Approximately 40 percent of this saving is attributed to the elimination of ballute testing and 60 percent to the elimination of the Mars heat shield qualification requirements. This saving must be weighed against the additional spacecraft development and operational costs incurred by an increase of 3000 to 10, 000 pounds (1360 to 4540 kg) in MEM weight. This tradeoff is beyond the scope of the present study. However, if the increased MEM weights do not require a significant number of additional Earth-orbital launch vehicles, this saving will be significant.

DEVELOPMENT PROGRAM COST SUMMARY

COST ELEMENT	PROGRAM	
	RECOMMENDED PROGRAM	BALLUTES & MARS ENTRY HEAT SHIELD
STRUCTURE	.802	1.052
ELECTRICAL POWER	.142	.188
COMMUNICATIONS AND INSTRUMENTATION	.172	.225
REACTION CONTROL	.115	.150
GUIDANCE AND CONTROLS	.200	.263
CREW SYSTEMS AND LIFE SUPPORT	.215	.282
PROPULSION	.530	.616
LANDING GEAR	.086	.113
BALLUTES	----	.363
GSE, SPARES AND OFF-SITE SUPPORT	.443	.582
SYSTEMS ENGINEERING & INTEGRATION	.357	.470
MEM DEVELOPMENT PROGRAM HARDWARE COSTS	$3.062 B	$4.304
LOGISTICS SUPPORT VEHICLES (3)	.146	.146
FACILITIES	.038	.038
LAUNCH OPERATIONS	.454	.572
PROGRAM MANAGEMENT & SUPPORT	.370	.510
TOTAL PROGRAM COSTS	**$4.07 B**	**$5.57 B**

BASIS: 4-MAN/30-DAY
LOW L/D CONFIGURATION
ELLIPTICAL ORBIT MISSION

RECOMMENDED MEM PROGRAM

The recommendations and results of the study are summarized as follows:

a. A low L/D MEM configuration is recommended because it affords lower weights and costs. A 4-man 30-day elliptical orbit mission (e = 0.9) was selected for the design mission because it imposes the most stringent requirements. This vehicle can be off-loaded to perform less demanding missions (e.g., from lower orbits, smaller crews, or shorter stay times).

b. A heat shield capable of entry from Earth orbit is recommended because it results in lower program costs and desensitizes the MEM design to the Mars entry environment. Retro-propulsive descent is recommended because it affords operational simplicity and also significantly reduced development costs.

c. The MEM test and qualification program will consist of ground-based, Earth suborbital and orbital tests. These tests can be designed to simulate all the Mars mission operational phases and, therefore, an unmanned flight to Mars, either with a full or subscale vehicle, is not required.

d. To achieve a Mars landing in 1982, the Phase D hardware acquisition program must begin by 1974. The Mars atmosphere model must be validated by 1978.

e. Program costs for the recommended MEM development program, up to but not including the first MEM delivered to the spacecraft integration contractor, will be between $4.1 and $5.0 billion, depending upon the increases in complexity and weight experienced during the development program.

RECOMMENDED MEM PROGRAM

MEM DESIGN

LOW L/D CONFIGURATION

CAPABLE OF ELLIPTICAL ORBIT MISSION

4 MEN/30 DAYS, ASCENT ΔV = 20,350 FPS
OFF-LOAD FOR LESS DEMANDING MISSIONS

EARTH ORBITAL ENTRY HEAT SHIELD

RETRO ONLY DECELERATION (NO BALLUTES)

TEST AND QUALIFICATION

GROUND AND EARTH ORBITAL TESTS

NO UNMANNED MARS FLIGHT TESTS

1974 PHASE D GO-AHEAD FOR 1982 LANDING

1978 ATMOSPHERIC VERIFICATION

COSTS \$4.1 - \$5.0 $\bar{B}$